AD 734236

DOCUMENT CONTROL DATA - R & D

(Security classification of title, body of abstract and indexing annotation must be entered when the overall report is classified)

1. ORIGINATING ACTIVITY (Corporate author)	2a. REPORT SECURITY CLASSIFICATION
Vertol Division The Boeing Company P. O. Box 16858, Philadelphia, PA 19142	Unclassified
	2b. GROUP N/A

3. REPORT TITLE

Four Prop Tilt Wing with Cyclic Pitch Propellers: Results of Full Span Wind Tunnel Test/Phase II

4. DESCRIPTIVE NOTES (Type of report and inclusive dates)

Contractor Test Report (December 1970)

5. AUTHOR(S) (First name, middle initial, last name)

Kolesar, Charles

6. REPORT DATE	7a. TOTAL NO. OF PAGES	7b. NO. OF REFS
June 1971	294	2

8a. CONTRACT OR GRANT NO.	9a. ORIGINATOR'S REPORT NUMBER(S)
F33615-70-C-1000	Boeing Vertol Document
b. PROJECT NO. 698BT	D170-10039-1
c. Task Area Number: 02	9b. OTHER REPORT NO(S) (Any other numbers that may be assigned this report)
d. Work Unit Number: 005	AFFDL TR 71-91, Reference 4

10. DISTRIBUTION STATEMENT

Approved for public release; distribution unlimited.

11. SUPPLEMENTARY NOTES	12. SPONSORING MILITARY ACTIVITY
	Air Force Flight Dynamics Laboratory Wright-Patterson AFB, Ohio 45433

13. ABSTRACT

This report presents the results of wind tunnel test BVWT 067, the Phase II test of a two phase test program, performed in the Boeing-Vertol V/STOL wind tunnel on a powered four prop tilt wing full span model equipped with cyclic pitch propellers for longitudinal control. Items evaluated through transitional flight, include cyclic pitch effectiveness, descent performance with cyclic pitch inputs, the effect of cyclic on longitudinal and lateral/directional stability plus the influence of cyclic action on the effectiveness of the aircraft surface controls (stabilizer for longitudinal trim and differentially deflected flaps/spoiler for roll/yaw control).

An in-ground effect investigation with a moving ground plane was also conducted to establish the influence of the ground on cyclic pitch effectiveness and on stability characteristics with cyclic pitch inputs.

14 KEY WORDS	LINK A		LINK B		LINK C	
	ROLE	WT	ROLE	WT	ROLE	WT
Cyclic Pitch Propellers						
Descent Capability						
Double Slotted Flaps						
Hover						
In-Ground Effect						
Lateral/Directional Stability and Control						
Longitudinal Stability and Control						
Propeller Hub Pitching Moment						
Propeller Normal Force						
Slats						
Tilt Wing Aircraft						
Transitional Flight						

II

THE **BOEING** COMPANY

VERTOL DIVISION · PHILADELPHIA, PENNSYLVANIA

CODE IDENT. NO. 77272

NUMBER D170-10039-1

TITLE FOUR PROP TILT WING WITH CYCLIC PITCH

PROPELLERS: RESULTS OF FULL SPAN

WIND TUNNEL TEST/PHASE II

ORIGINAL RELEASE DATE JUNE 1971 . FOR THE RELEASE DATE OF
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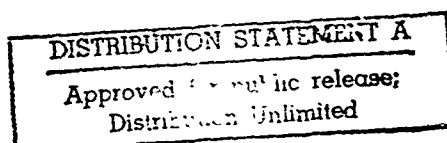
MODEL _____ CONTRACT F33615-70-C-1000

ISSUE NO. __ _____ ISSUED TO: _____

PREPARED BY _Charles E. Kolesar_ DATE 6/17/71

APPROVED BY _Charles E. Kolesar_ DATE 6/17/71
C. E. Kolesar

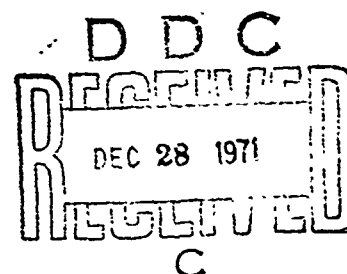APPROVED BY _____ DATE 6/30/71
K.B. Gilmore, Mgr., V/STOL Technology

APPROVED BY _____ DATE 6/30/71
W.L. Lapinski, Program Manager

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FORM 46238 (7/67)

VII

SHEET v

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SHEET vi

REVISIONS			
LTR	DESCRIPTION	DATE	APPROVAL

ABSTRACT

This report presents the results of wind tunnel test BVWT 067,
the Phase II test of a two phase test program, performed in
the Boeing-Vertol V/STOL wind tunnel on a powered four prop
tilt wing full span model equipped wtih cyclic pitch propellers
for longitudinal control. Items evaluated through transitional
flight, include cyclic pitch effectiveness, descent performance
with cyclic pitch inputs, the effect of cyclic on longitudinal
ana lateral/directional stability plus the influence of cyclic
action on the effectiveness of the aircraft surface controls
(stabilizer for longitudinal trim and differentially deflected
flaps/spoiler for roll/yaw control).

An in-ground effect investigation with a moving ground plane
was also conducted to establish the influence of the ground on
cyclic pitch effectiveness ana on stability characteristics
with cyclic pitch inputs.

KEY WORDS

Cyclic Pitch Propellers

Descent Capability

Double Slotted Flaps

Hover

In-Ground Effect

Lateral/Directional Stability and Control

Longitudinal Stability and Control

Propeller Hub Pitching Moment

Propeller Normal Force

Slats

Tilt Wing Aircraft

Transitional Flight

NOMENCLATURE

The following nomenclature was used for Model VR068Q in BVWT 067. Additional nomenclature is included in the Data Reduction section of this report.

<u>Symbol</u>

A_p	Propeller disc area	ft.^{2}
α_F	Fuselage angle of attack relative to freestream	degrees
$\alpha_{W_{EFF}}$	Effective wing angle of attack	degrees
B_1	Fuselage	
b	Wing span	ft.
c	Mean aerodynamic wing chord	ft.
γ	Cyclic angle (positive $\sim$ nose down pitching moment)	degrees
D	Propeller diameter	ft.
δ_F	Flap angle	degrees
F_1	Wing fence configuration	
f_1	Basic double slotted flaps	
$\theta_{.75}$	Propeller blade pitch angle at .75R	degrees
H_1	Horizontal tail	
h	Height of outboard propeller plane to ground plane in hover	inches
i_W	Wing incidence angle	degrees
J	Propeller advance ratio, $\dfrac{V}{nD}$	
L	Aircraft lift	lbs.
M	Aircraft pitching moment (positive $\sim$ nose up)	ft.lbs.

2

N_1	Nacelle configuration	
n	Propeller rotational speed	rps
P_1	Collective hubs	
P_2	Cyclic hubs	
Q	Shaft torque	ft-lbs
Q_1	Basic slat configuration	
q	Freestream dynamic pressure	lbs/ft^2
q_s	Slipstream dynamic pressure, $q+T/A_p$	lbs/ft^2
R	Propeller blade radius	ft.
r	Radial station along blade	ft.
ρ	Density	$slugs/ft^3$
S	Wing area	$ft.^2$
ι	Horizontal stabilizer incidence relative to waterline	degrees
T	Propeller thrust	lbs.
T_j	Jet thrust from air motor	lbs.
V	Velocity	ft/sec.
V_F	Full scale aircraft velocity	knots
V_1	Vertical tail	
W_1	Wing	
X	Aircraft longitudinal force, positive forward	lbs.
$\dot{x}$	Longitudinal distance	ft.
z	Vertical distance	ft.

<u>Superscripts</u> - (Superscripts are in sequence, left wing
tip to fuselage centerline)

f^{60} Flap at 60°

$p^{1,1}$ Both propellers turning down inboard

$p^{1,2}$ Both propellers turning down between
nacelles

Q^{10} Slat setting, see notes

Q^* Slat setting, see notes.
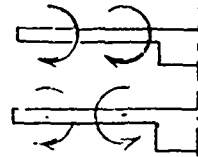
NOTES: 1)   According to the notation used, slat setting
$Q^{10,10,10}*$ indicated that all slat segments
inboard of the wing tip were set at Q^{10} position
with the exception of the segment inboard of
the inboard nacelle which was set at Q^* position.

4

TABLE OF CONTENTS

TABLE OF CONTENTS

1.0 INTRODUCTION

Wind tunnel test BVWT 067 was conducted in the 20ft. x 20ft. test section of the Boeing-Vertol V/STOL wind tunnel on a full span (9.04ft.) model of the Model 170 four propeller tilt wing aircraft that utilizes cyclic pitch propellers for low speed longitudinal control. This test, which covered the month of December 1970, was the Phase II test of a two phase test program performed on sting mounted VRO68Q model. Results of the Phase I test, conducted in July/August 1970, are reported in Reference 1.

This air motor powered model with 2.14ft. diameter propellers, incorporated an internal six copmonent stra. n gage fuselage balance and a six component strain gage balance mounted in each nacelle between the cyclic pitch propeller hub assembly and the air motor. A similar propeller/cyclic pitch hub/ nacelle balance/air motor assembly was tested as an isolated propeller during May 1970 as a part of the contract (Reference 2).

The primary objectives of this Phase II test were:

a. Yaw Control in Hover

Complete the hover yaw control testing with an investigation of the yaw control capability of a combined flap/spoiler configuration, down flap deflection on one wing and spoiler deflection on the opposite wing, plus a configuration with differential flaps and spoilers.

b. Low Speed Descent Performance

Complete the investigation started in the Phase I test evaluating the effect of cyclic pitch on descent capability.

c. Cyclic Pitch Effectiveness in Transition

Determine the longitudinal control capability of the cyclic propellers at points through transition with selected combinations of wing tilt angle and flap angle.

d. Longitudinal Stability with Cyclic Pitch Inputs

Establish the influence of cyclic pitch on both the horizontal tail-off and tail-on longitudinal stability characteristics in the transition regime.

7

e. Basic Lateral/Directional Stability in Transition

    Determine the vertical tail effectiveness and
    tail-on lateral/directional stability character-
    istics at selected combinations of wing incidence,
    flap angle and slipstream thrust coefficient.

f. Lateral/Directional Stability with Cyclic Pitch Inputs

    Establish the effect of cyclic pitch on the tail-
    on lateral/directional stability characteristics
    at points through transition.

g. Aircraft Surface Control Power Coupled with Cyclic Pitch

    Determine whether cyclic pitch inputs exert an
    influence on the stabilizer control effectiveness
    in transitional flight and on the flaps and spoilers
    used for roll/yaw control in transition (depending
    upon the wing tilt) and for yaw control in hover.

h. Effect of Ground Proximity on Longitudinal Characteristics
    and Cyclic Pitch Control

    Determine the resultant changes in transitional
    longitudinal characteristics due to an in-ground
    effect condition and its influence on the cyclic
    pitch effectiveness.

i. Effect of Ground Proximity on Lateral/Directional
    Stability with and without Cyclic Pitch Inputs

    Establish whether ground effect influences the
    basic lateral/directional characteristics of the
    aircraft as determined for transition and evaluate
    the changes resulting from cyclic pitch application.
    Investigate for ground recirculation.

Four prop Model VRO68Q utilizes a 9.23 aspect ratio tapered
wing with a straight leading edge, propellers overlapped 7% in
diameter, full span slats and full span large chord double
slotted flaps that incorporate a movable fore flap which "nests"
when the flap is retracted. Longitudinal and vertical location
of the propeller hub centerlines with respect to the wing lead-
ing edge were chosen to maximize descent capability, using as
a basis the data acquired in mid-1969 during a Boeing-Vertol
wind tunnel test of a semispan four prop tilt wing model, the
primary objective of which was to investigate the effect of
propeller hub location on descent performance with overlapped
propellers. The slat and double slotted flap configurations

8

used on Model VRO68Q were also established from data acquired
during the 1969 Boeing-Vertol four prop tilt wing wind tunnel
test program, that included investigations of single vs double
slotted flaps and full span slats vs Kruger leading edge flaps
for the purpose of maximizing descent performance. Two wing
fences per wing, one at the fuselage side and the other, 18%
of a propeller diameter outboard, were used to contain the
stall occurring on the wing center section. The swept vertical
tail incorporated a tail volume of .083 and the horizontal tail
mounted high on the vertical had a tail volume coefficient of
1.33.

2.0 MODEL DESCRIPTION AND INSTALLATION

The general arrangement and geometry of full span Model VRO68Q and wind tunnel installation details are presented in this section. Figure 1 is a photograph of this model as installed in the Boeing-Vertol V/STOL wind tunnel for the subject test.

2.1 WING GEOMETRY (See Figure 2)

The model utilizes a tapered wing with the following geometry:

Span	9.036 ft.
Root chord	1.263 ft.
Tip chord	0.696 ft.
Mean Aerodynamic Chord (MAC)	1.007 ft.
Taper ratio	0.551
Area	8.850 ft.^{2}
Aspect ratio	9.225
Wing 1/4c sweep	1.6° fwd.
Dihedral	0°
Wing pivot position	
X-Axis	42.56% MAC aft of wing L.E.
Z-Axis	11.67% MAC below w.c.p.
Basic Wing Sections	
Root	NACA 64$_4$221
Tip (actually 1.047 b/2)	NACA 64$_2$215
Inboard Nacelle Wing Chord/Prop Diameter	0.492
Outboard Nacelle Wing Chord/Prop Diameter	0.376
Slats (from wing tip to body centerline)	15% basic wing chord

Double Slotted Flaps   39% basic wing
                                        chord (when re-
                                        tracted)
                                        23.5% chord Fowler
                                        action

Slats

The Model VRO68Q tapered wing incorporated a 15% chord full
span leading edge slat of the design illustrated in Figure 3.
Figures 4 and 5 show the slat positioned at tip and root sec-
tions, respectively. The slat, which extended spanwise across
the fuselage to the body centerline (wing root), was attached
to the basic wing leading edge with preset brackets,and were
arranged in nine spanwise segments (wing tip to wing tip).
One segment per wing extended from the wing tip to the out-
board nacelle; the slat between the nacelles was divided into
two equal span segments; one segment extended from the side of
the inboard nacelle to the fuselage; and finally, one segment
covered the entire width of the fuselage. This arrangement
enabled the slats to be set differentially according to the
direction of rotation of the propeller blades in front of each
slat segment.

The slat angle, gap and trailing edge location used for this
test were based on previous Boeing-Vertol tilt wing testing that
was conducted for the purpose of maximizing descent capability.
In general, the Q^{10} slat setting shown in Figures 3 and 4 was
used behind a "down-going" propeller blade and the Q* setting
was used behind an "up-going' propeller blade.

Flaps

The large chord double slotted flaps used in test BVWT 061 are
shown in Figures 6 and 7, which present the arrangement of the
flaps for the 40°, 50°, and 60° deflections. Figures 6 and 7
depict tip and root sections, respectively. Nacelle length
precluded extending the flaps behind the nacelles, thus split-
ting the flaps into three spanwise segments: outboard, mid-
span, and inboard. The lengths of these segments were as
follows:

Flap Segment	Length
Outboard	STA 54.215" (tip) to STA 45.789"
Midspan	STA 41.789" to STA 21.875"
Inboard	STA 17.875" to STA 7.145"(side of body)

The flaps were 39% chord in the retracted position. In this
position, the fore flap "nests" against the main flap as shown
in Figure 8. During the initial portion of the flap extension,

11

the "nested" fore flap/main flap assembly is moved aft 23.5%
of the basic wing chord. This value r)resents the Fowler
action. For the first 20° of flap deflection, the fore flap
remains "nested" as shown in Figure 9. As the flap is deflec-
ted to a higher angle, the main flap moves away from the fore
flap, resulting in an extended flap chord of 49% at the 40°
flap angle. The geometric relationship between the fore flap
and main flap was held constant between 40° and 60° flap angle
(the maximum angle tested). Gaps and locations used for the
double slotted flaps were determined from previous test data.

Yaw control in hover is provided by flap down-travel on one
wing, plus spoiler and flap up-travel on the opposite wing.
The arrangement of the flap in the up-travel position for hover
is illustrated in Figure 9.
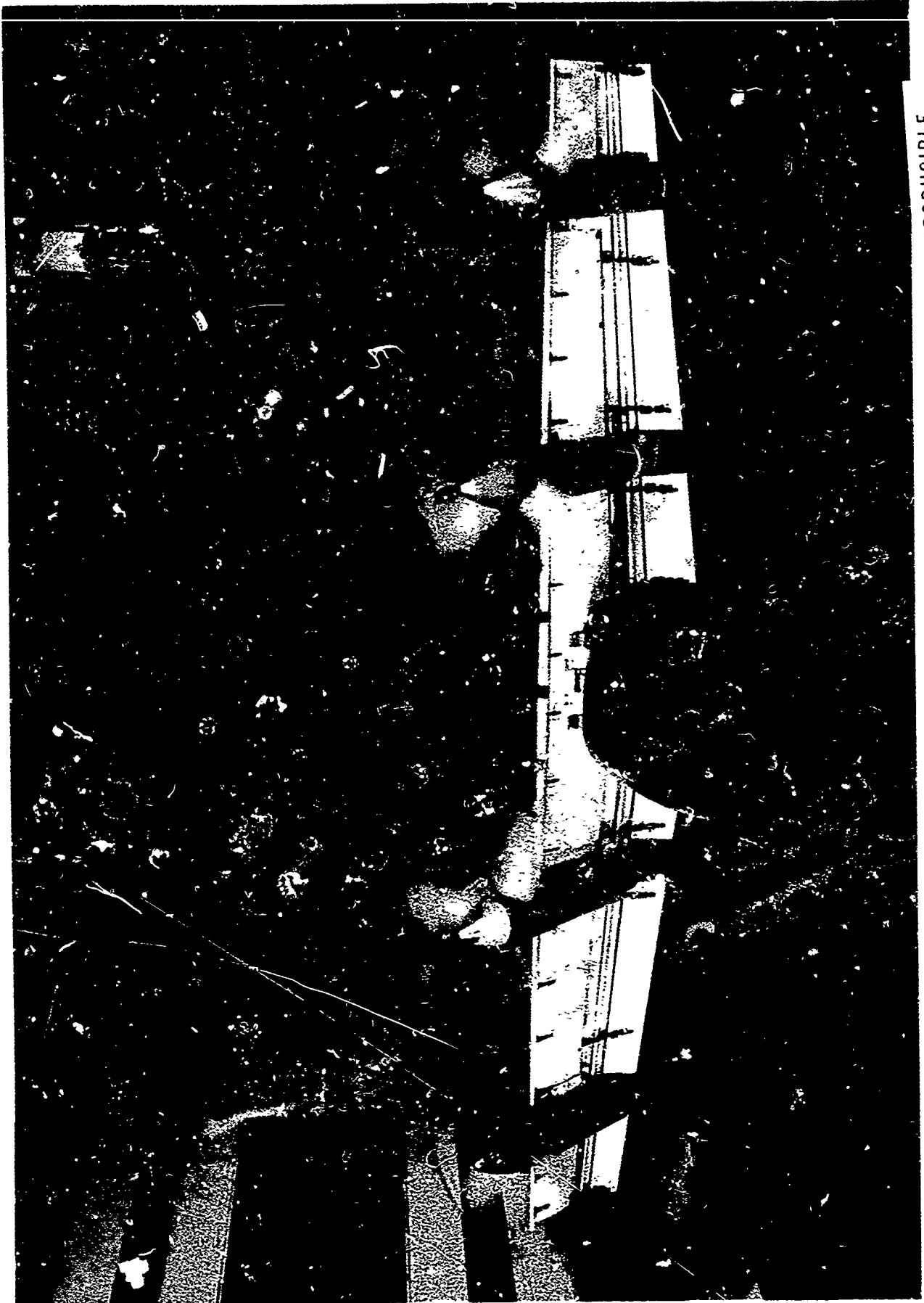
Figure 1

FULL SPAN MODEL VRO68Q
INSTALLATION IN BOEING-VERTOL
V/STOL WIND TUNNEL WITH MOVING GROUND BELT

NOTES:
1. Wing Area = 8.85ft²
2. Prop Area = 3.607ft²
3. Aspect Ratio = 9.225
4. Taper Ratio = 0.551
5. F defines wing fence
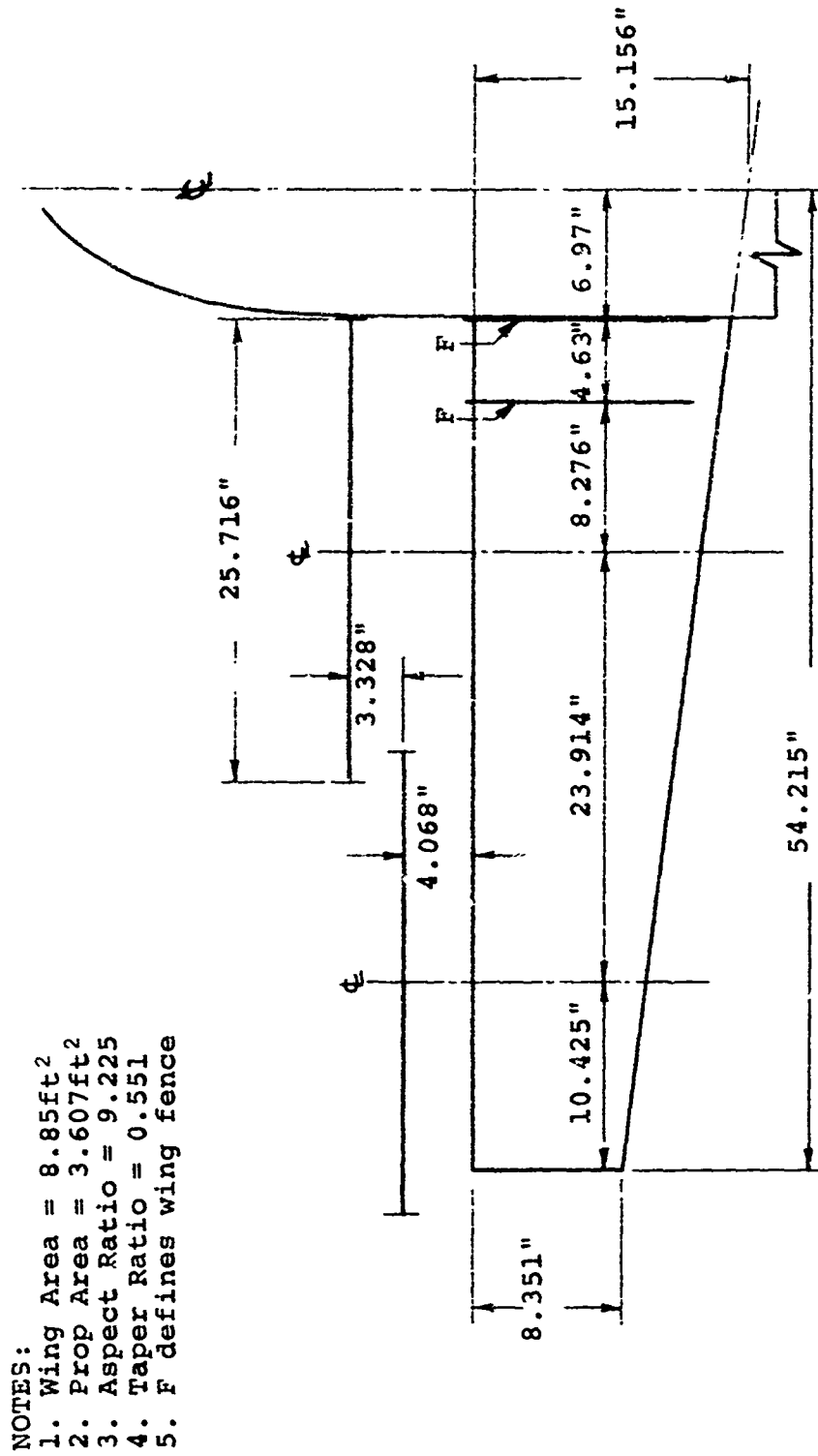
MODEL VRO68Q
BASIC WING GEOMETRY
AND PROPELLER LOCATION
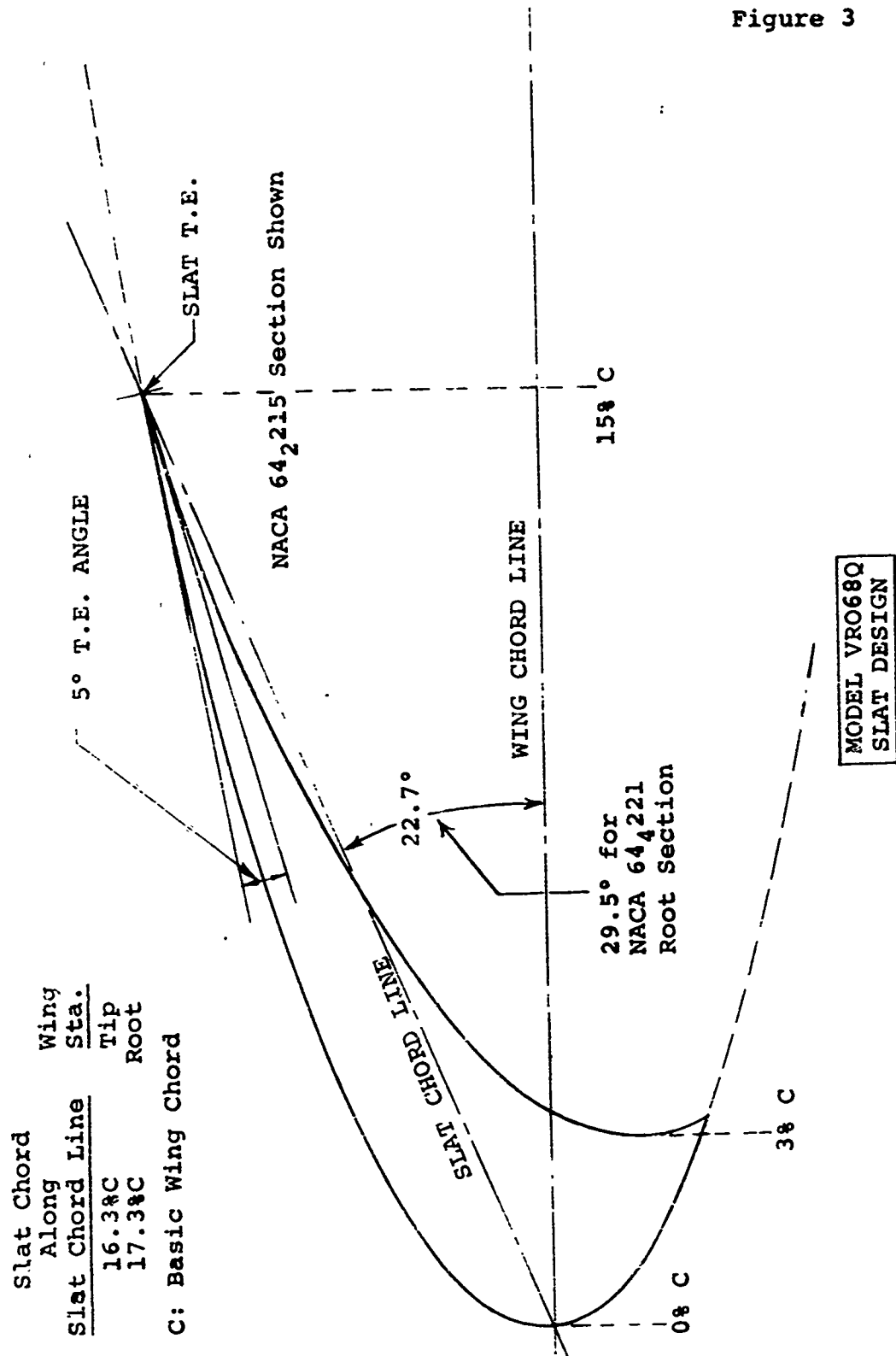
15.156"

6.97"

4.63"

8.276"

25.716"

3.328"

23.914"

4.068"

54.215"

10.425"

8.351"

F

F

14

SLAT T.E.

NACA 64₂215' Section Shown

5° T.E. ANGLE

WING CHORD LINE

15% C

22.7°

29.5° for
NACA 64₄221
Root Section

08 C

3% C

SLAT CHORD LINE

Slat Chord
Along
Slat Chord Line	Wing Sta.
	Tip
16.3%C	Root
17.3%C

C: Basic Wing Chord

MODEL VRO68Q
SLAT DESIGN

Figure 4

0.10%c

51.1°
Wing
Chord Plane
(wcp)

1.32%c*

Slat Chord

SLAT POSITION Q_1^*

*Increased to 1.50%c

0.65%c

45.1°
Wing Chord Plane
(wcp)

1.46%c

Slat Chord
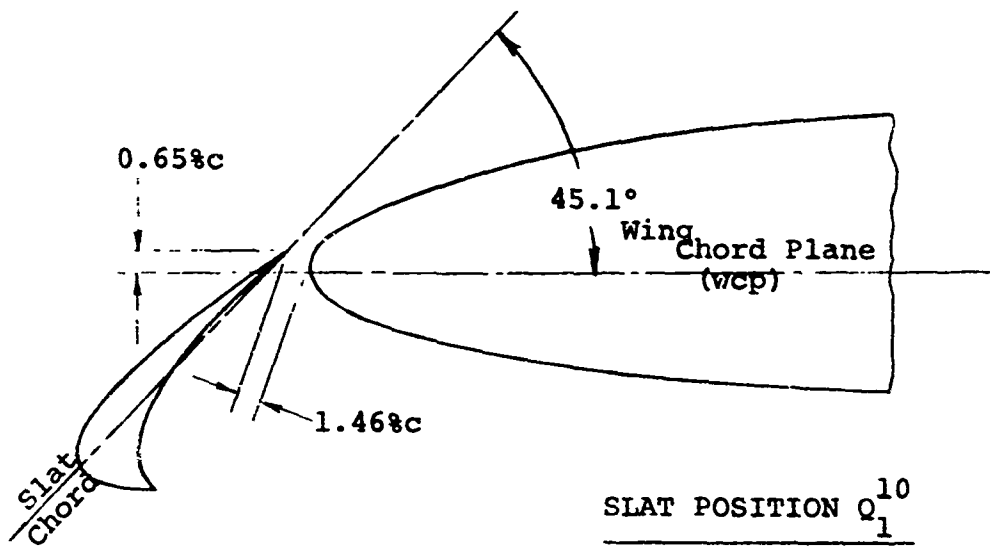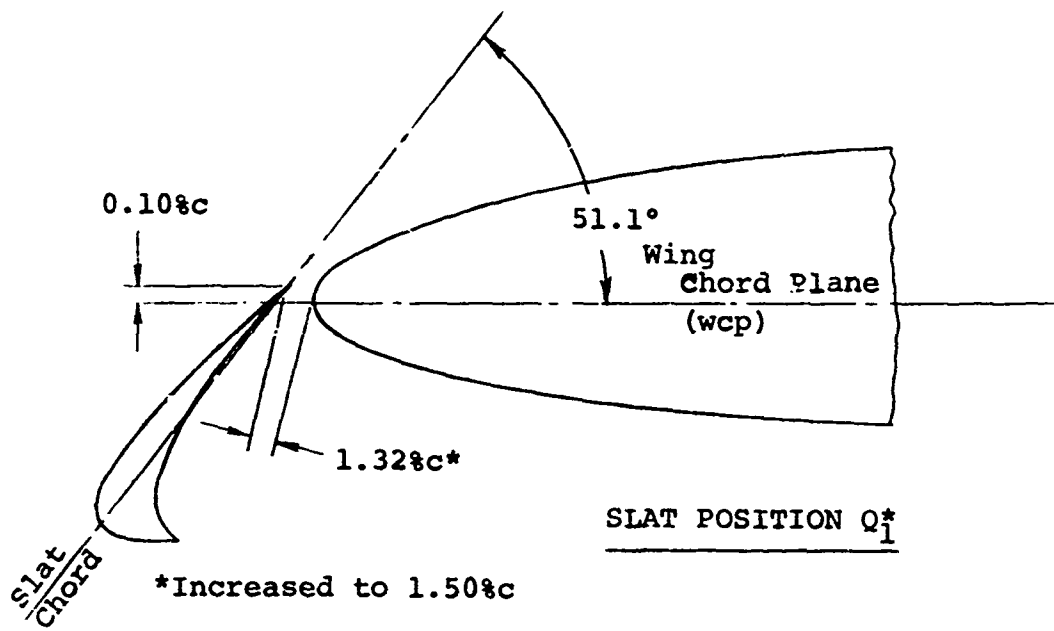
SLAT POSITION Q_1^{10}

MODEL VRO68Q
SLAT POSITIONS
TIP SECTION

16

Figure 5

0.90%c

51.1°

Wing Chord Plane
(wcp)

1.62%c

Slat Chord

SLAT POSITION Q_1^*

1.65%c

45.1°

Wing Chord Plane
(wcp)

1.77%c

Slat Chord

SLAT POSITION Q_1^{10}

MODEL VR068Q
SLAT POSITIONS
ROOT SECTION

17

MODEL VR068Q
DOUBLE SLOTTED FLAP GEOMETRY
40°,50°, & 60° POSITIONS
TIP SECTION

Spoiler T.E. (85%c)

Fore Flap Chord Line

Wing Chord Line

−.129"R

30°*

Main Flap Chord Line

.043"

X_1

G_1

Flap Angle

4.5°

Main Flap Chord Line

Main Flap Chord: 34.6%c
Fore Flap Chord: 15.1%c

Flap Angle	G_1		X_1	
60°	1.518c	.126"	.258c	.021"
50°	1.518c	.126"	.75c	.063"
40°	1.518	.126"	1.63%c	.136"

c: Basic Wing Chord

*30° angle between main flap chord line
and fore flap chord is for section from
outboard nacelle to wing tip. 32° angle
for section between nacelles.

18

MODEL VRO68Q
DOUBLE SLOTTED FLAP GEOMETRY
40°,50°, & 60° POSITIONS
ROOT SECTION
(BODY ℄)

Spoiler T.E.(85%c)

Wing Chord Line

Fore Flap Chord Line

.242"R

Main Flap Chord

34°*

.095"

X_1

G_1

Flap Angle

5.7°

Main Flap Chord
(Retracted)

Fore Flap Chord: 15.4%c
Main Flap Chord: 34.6%c
C: Basic Wing Chord

Flap Angle	G_1		X_1	
60°	1.50%c	.227"	.528c	.079"
50°	1.50%c	.227"	1.178c	.178"
40°	1.76%c	.267"	1.658c	.250"
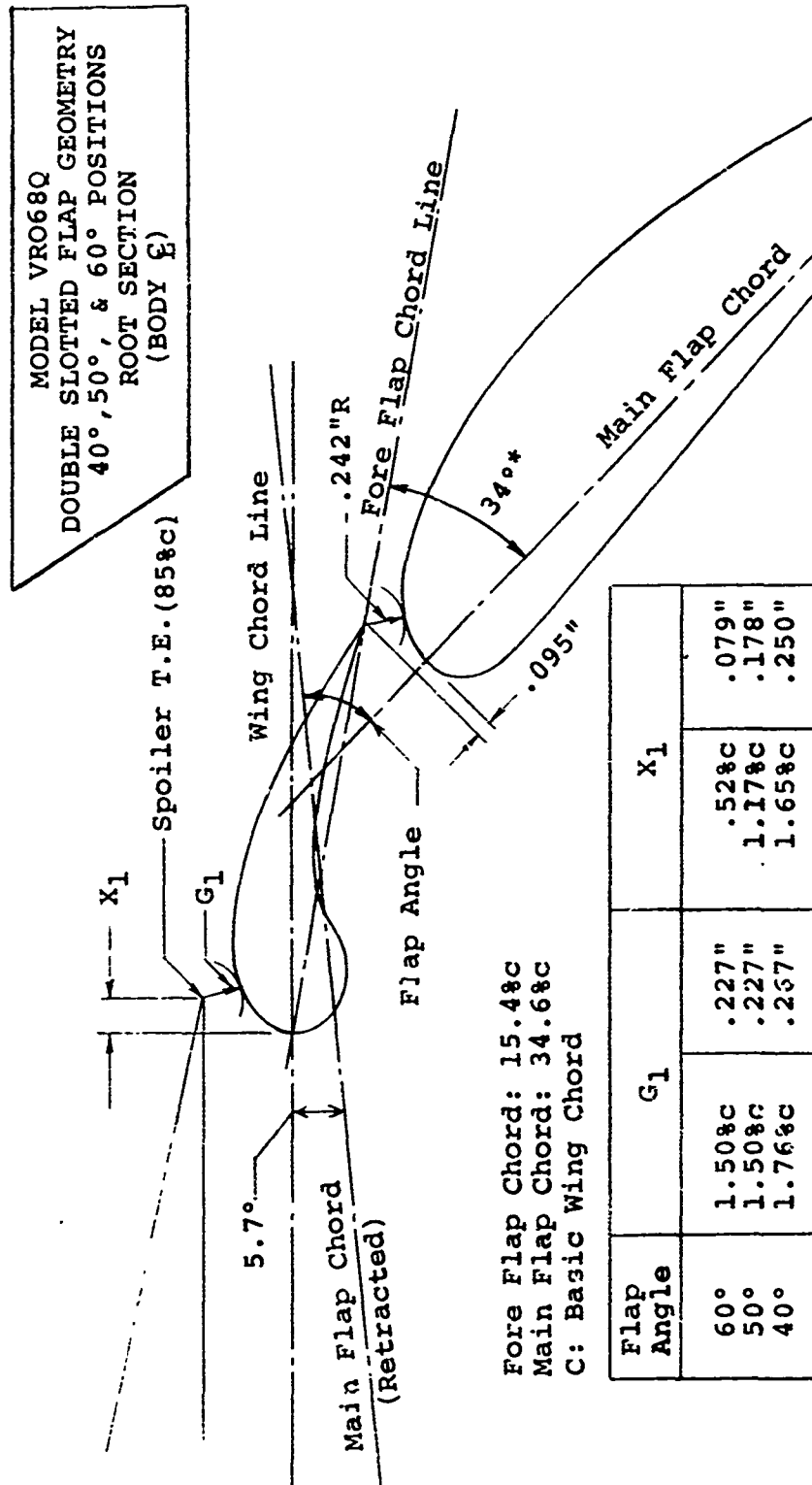
*34° angle between main flap chord line and
fore flap chord line is for section from
root to inboard nacelle. 32° angle for section
between nacelles.

ROOT SECTION ~ BODY ℄

TIP SECTION
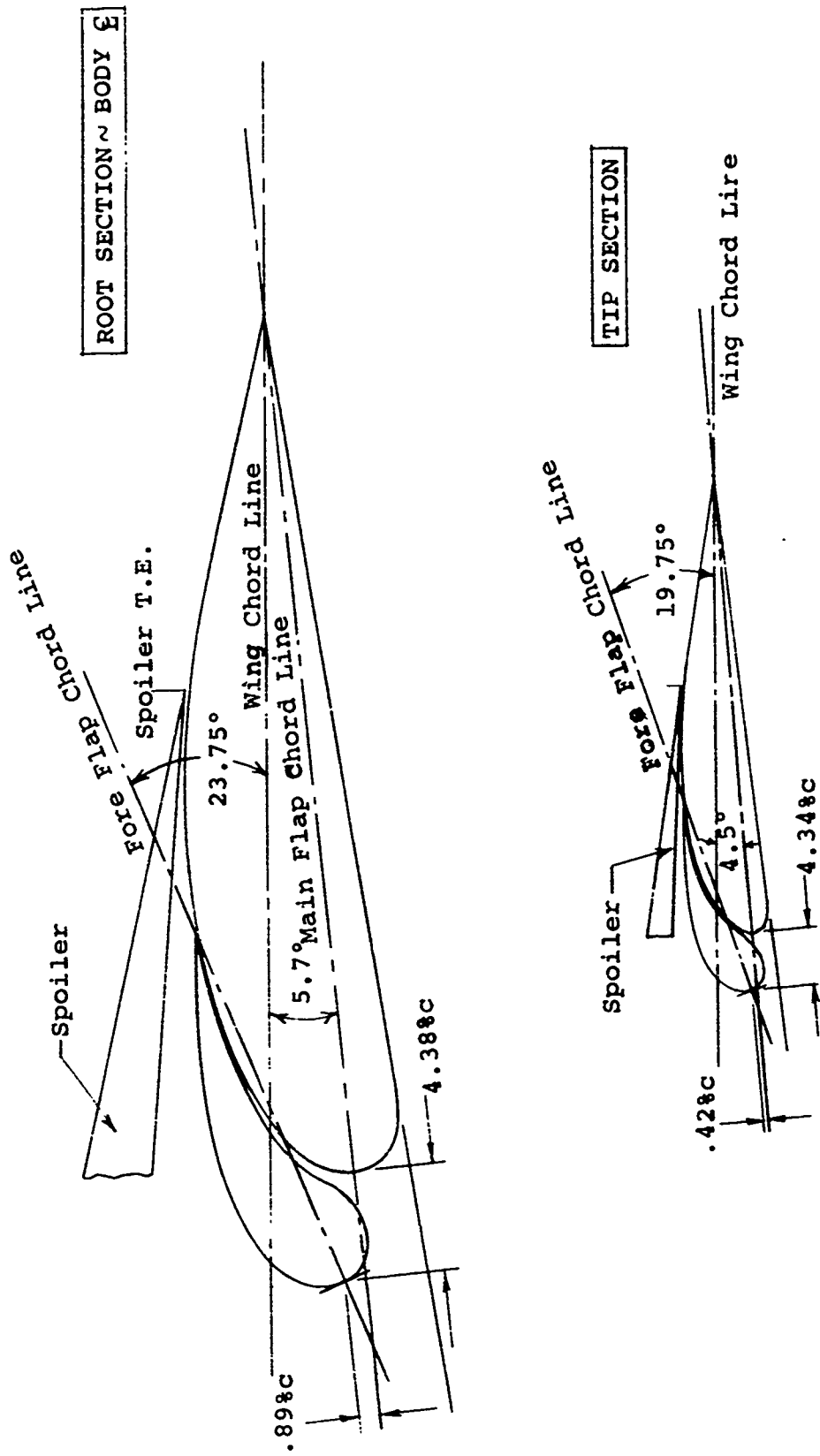
MODEL VRO68Q
DOUBLE SLOTTED FLAP GEOMETRY
RETRACTED POSITION

20

Figure 9

MODEL VRO68Q

DOUBLE SLOTTED FLAP GEOMETRY

20° AND UP-FLAP POSITIONS

Spoiler
T.E.

R=1.51%c (Tip) & 1.76%c (Root∿ Body ₵)

Wing Chord Line

Flap Chord Line

20°

Main Flap Chord Line

.54%c

20° FLAP POSITION

c: Local Wing Chord

UP-FLAP POSITION

Spoiler
T.E.

Main Flap Chord Line

-40°

T.E.

Gap
.17%c

Wing Chord Line

Flap Chord Line

.64%c

.39%c

2.2 PROPELLER, HUB GEOMETRY
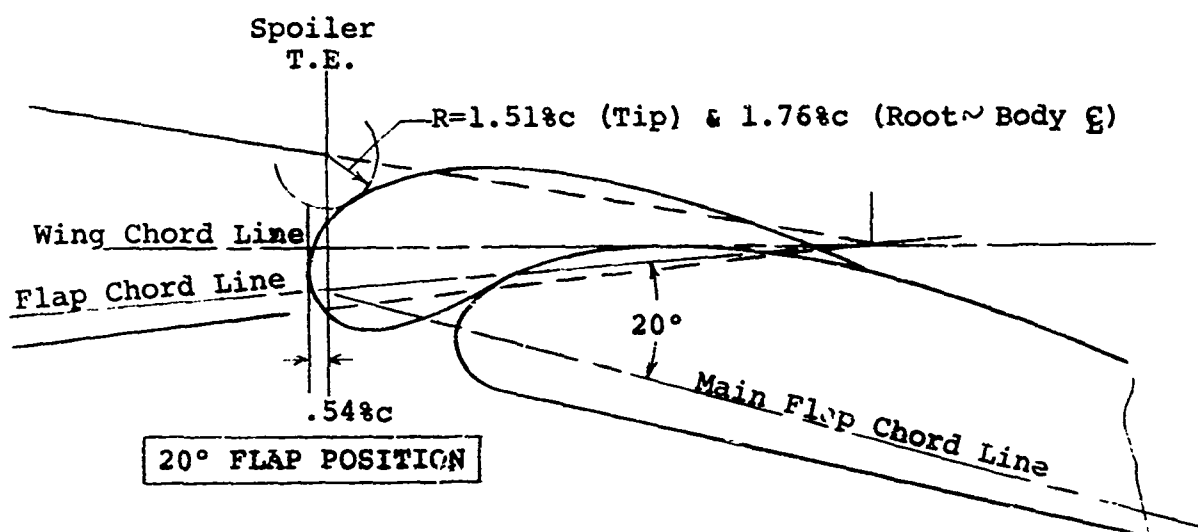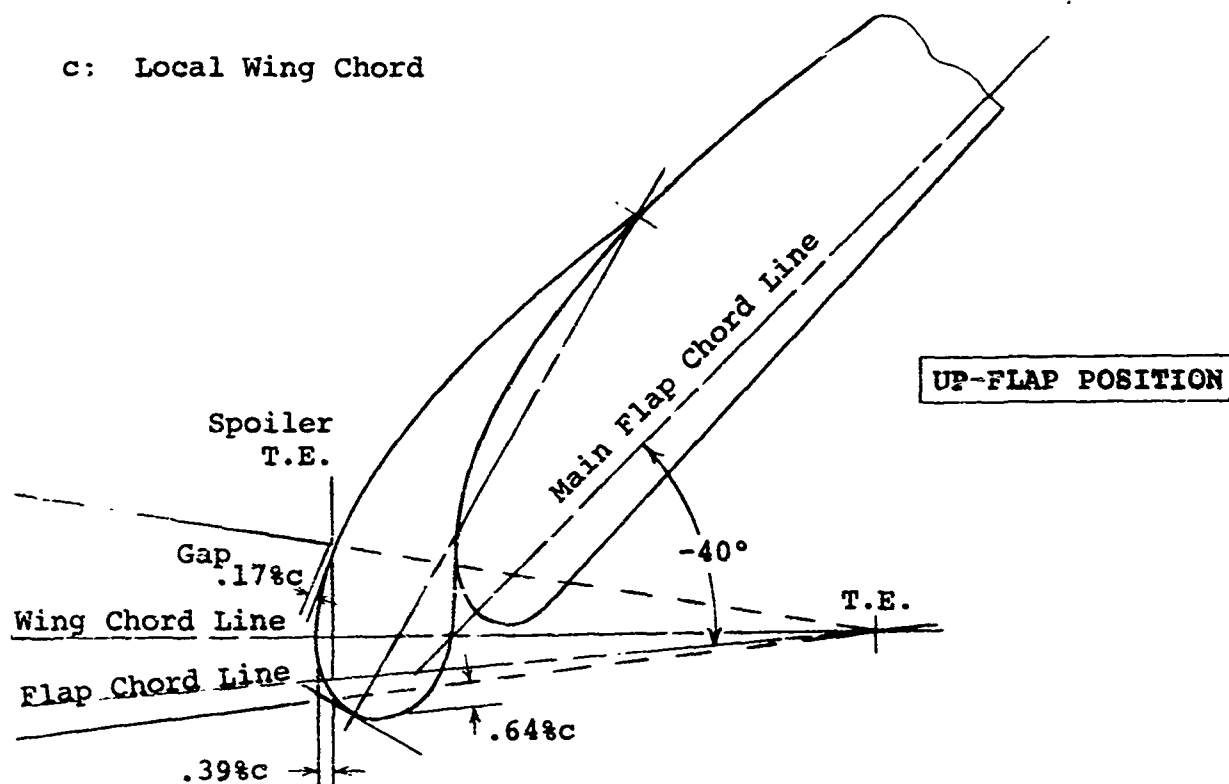
Geometric characteristics of the propeller blade used in this test are shown in Figure 10. The variation with radial station, of blade chord, design lift coefficient, thickness ratio and blade twist are presented in this plot. Figure 11 depicts the blade planform. Note that the blade pivots for manual collective settings about the 35% chord line.

Three-way collective hubs were used during the non-cyclic portion of the test. These were replaced with three-way cyclic hubs, using identical propeller blades, when cyclic pitch was required. Both propeller collective and cyclic angles were manually adjusted.

Figure 12 is a photograph of the 4.80 in. diameter cyclic hub. This hub employed a swashplate mounted on a cyclic stack fixed to the front of the six component nacelle balance. The outer annulus of the swashplate was driven by scissors mounted on the rear face of the hub. Cyclic pitch was applied to the blades through a set of pitch links. Elastomeric (Lamiflex) bearings were used in the hub for blade retention and blade angle motion.

Principal dimensional information and airfoil designations for the propellers are listed below:

Diameter	2.143 ft
Disc area	3.61 ft^2
Root chord (at .2r/R)	3.20 in.
Tip chord	2.32 in.
Root section	NACA 64A030
Tip section	NACA 64A306
Activity Factor	160 per blade
Overall Blade Twist	33.5°

For the Phase II test, the radial elastomeric bearings used in the cyclic hubs for previous testing were replaced with needle roller bearings to improve the hub life. This modification necessitated moving each blade in the hub, 0.125 in. outboard to allow for the greater depth of this type of bearing. Each propeller tip was clipped 0.125 in. to retain the original propeller diameter of 2.143 ft. The 1% reduction in blade area was considered negligible.

22

MODEL VRO68Q
PROPELLER
GEOMETRIC CHARACTERISTICS

NACA 64 SERIES (a=0.8 mod)

CHORD/DIAMETER

DESIGN LIFT COEFF

23-(e)

Figure 10

HUB ℄

ROOT

35% chord line

4.80"

3.20"

.20R
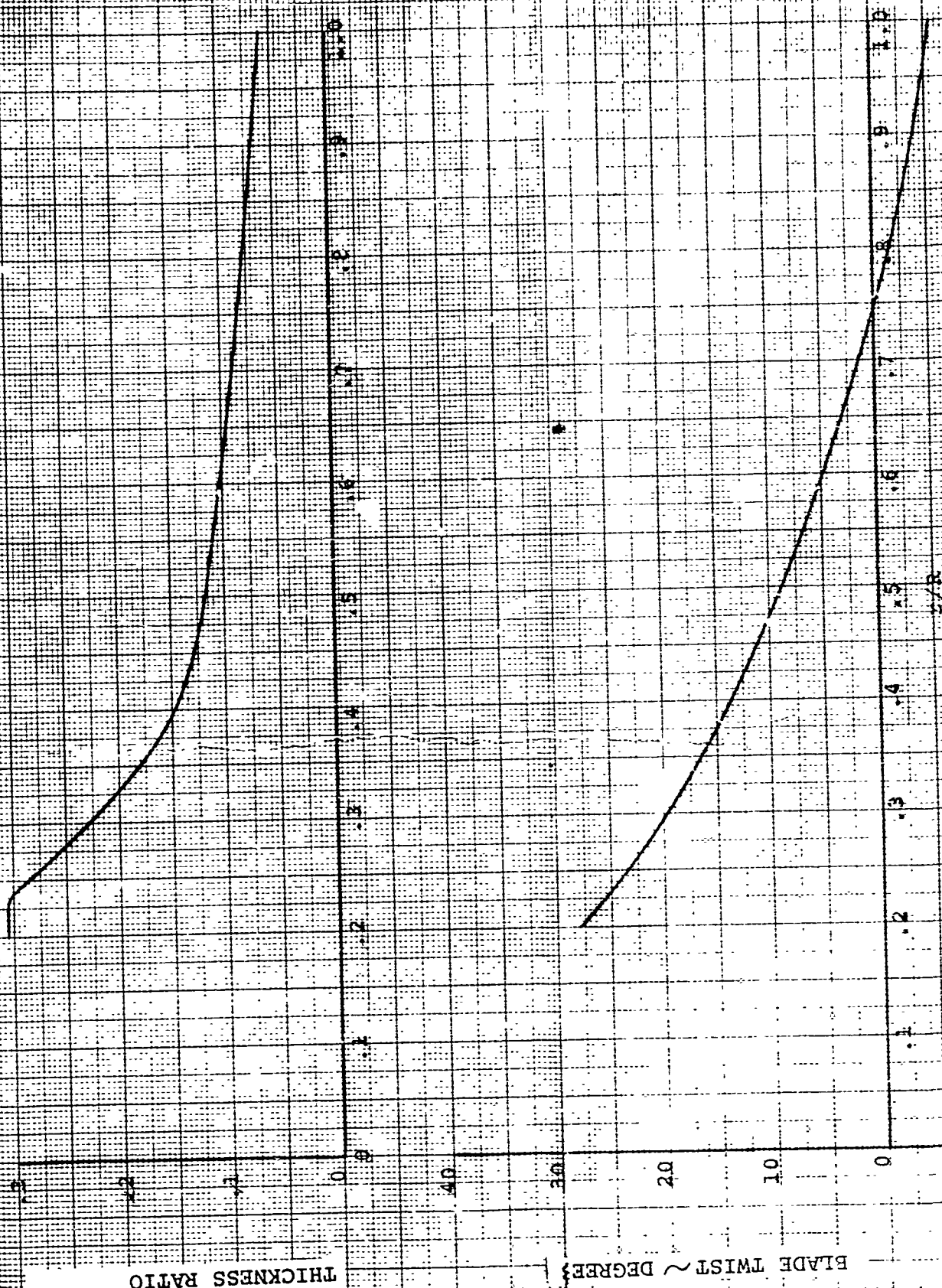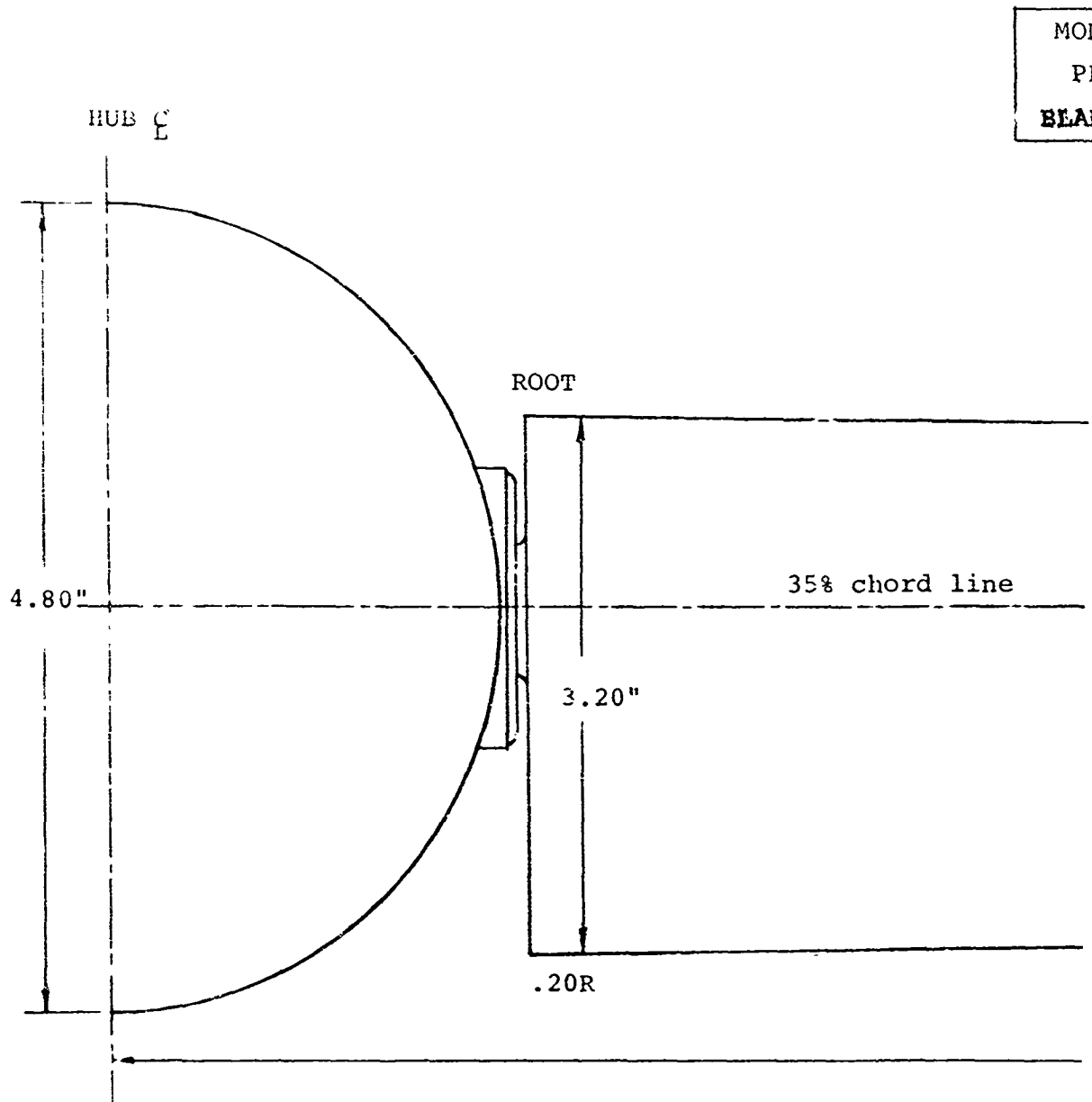
SCALE: FULL

NOTE: Actual Dimensions (Untwisted)

4

I B

Figure 11

L VRO68Q
PELLER
PLANFORM

TIP

2.78"

2.32"

.75 DATUM

1.0715 ft
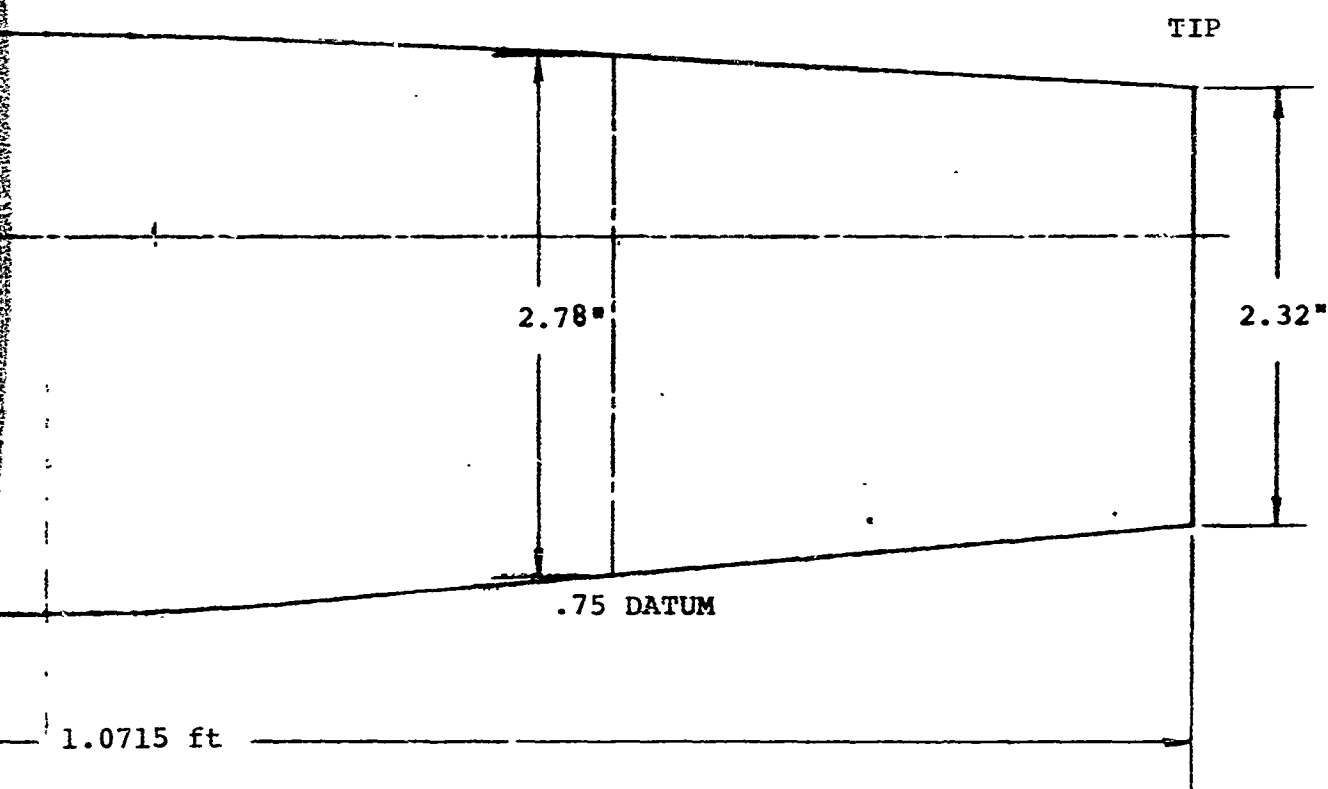
Details of Cyclic Hub Mechanism

25

2.3 NACELLE DESCRIPTION

A schematic drawing showing the arrangement of the propeller
hub, swashplate, slip ring, strain gage internal balance, and
air motor in the inboard nacelle along with the geometric re-
lationship of the wing with the nacelle and hub center is pre-
sented in Figure 13. Similar information is depicted for the
outboard nacelle in Figure 14. Not shown in the two sketches
is the flexible bellows coupling that joined the propeller
shaft to the air motor drive shaft. This coupling, located
1.875 in. aft of the nacelle balance center, isolated propeller
forces and moments.

The air motor power source, which utilizes a four stage tur-
bine, is designed to deliver 90 shaft horsepower at 9000 RPM.
At this design point, approximately 2 lb/sec of air flow is
required. Compressed air was individually ducted out to
the inboard and outboard air motors from the wing root thru
parallel 1.125 inch diameter air passages located internally
in the basic wing structure. A 90° bend was used to intro-
duce the air into the top of the inboard motor plenum in front
of the first stage turbine. The lack of sufficient wing spar
material at the outboard nacelle as a result of the relatively
high thrust line (See Figure 14) necessitated stopping the in-
ternal wing air passage short of the outboard nacelle and
angling the air into the outboard motor plenum (45° from ver-
tical) via a short air passage drilled into a wedge shaped
piece of material which was bolted to the lower surface of
the wing inboard of the outboard nacelle.

A diverging nozzle with eight straightening vanes for elimi-
nating exhaust swirl was attached to the rear end of each
motor.

The location of the propeller thrust line with respect to the
wing chord plane, and the prop plane location with respect to
the wing leading edge in terms of percentage of local wing
chord, can be determined from the information presented in
Figures 13 and 14. These values are listed below.

Prop Hub $\mathcal{C}$	Distance below wing chord plane	Distance ahead wing leading edge
Inboard	.218 z/c	.584 x/c
Outboard	.181 z/c	.421 x/c

The above ratios were established using data obtained during
a 1969 Boeing-Vertol semispan wind tunnel test of a four prop
tilt wing model. In this test, various prop hub centerline
locations with respect to the wing were evaluated to determine
the effect on descent capability.

26

2.750"

Wing Chord Line W.L.7.812

7.396"

Prop

Slip Ring

Air Motor

Balance

Balance
Moment
Ref.

5.375"

Spinner

Hub

Swashplate

Nozzle
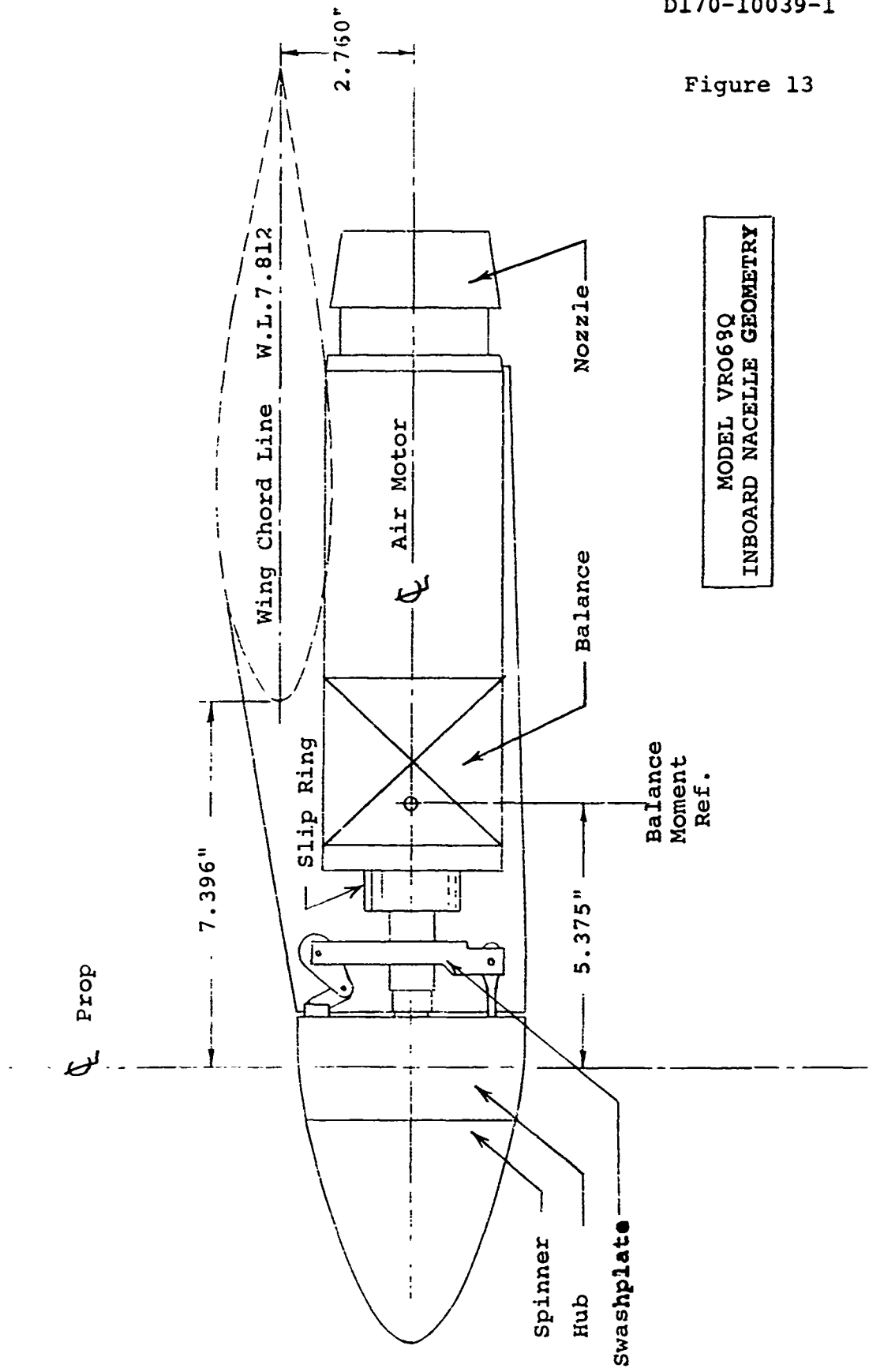
MODEL VRO69Q
INBOARD NACELLE GEOMETRY

27

D170-10039-1
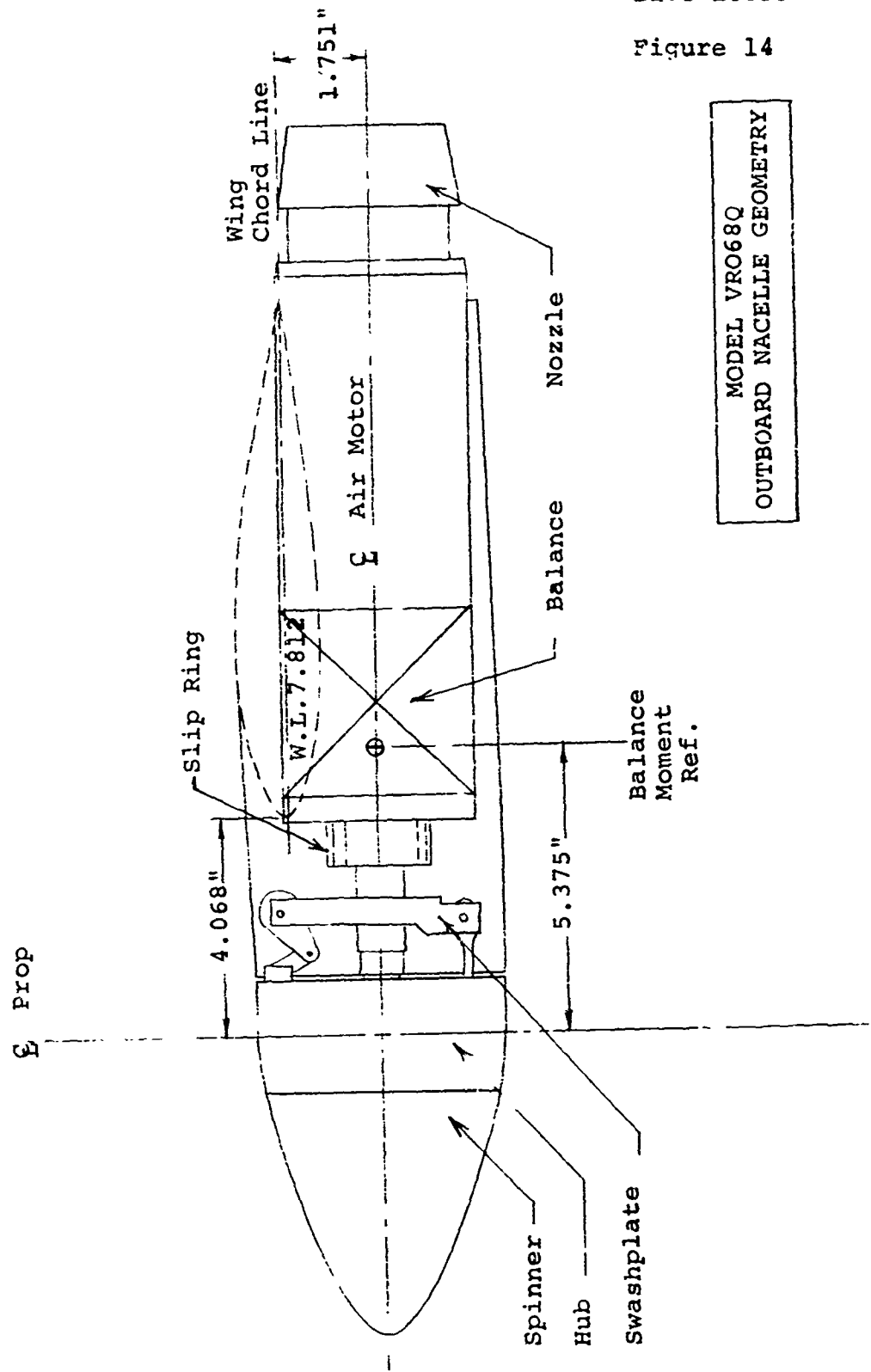
Figure 14

MODEL VRO68Q
OUTBOARD NACELLE GEOMETRY

Wing Chord Line

1.751"

Air Motor

Nozzle

Slip Ring

W.L.7.812

Balance

Balance Moment Ref.

4.068"

5.375"

Prop

Spinner

Hub

Swashplate

2.4 FUSELAGE GEOMETRY

The fuselage used on Model VRO68Q, as shown in Figure 15, had
a shape generally representative of a four propeller tilt wing
transport-type aircraft designed for rear ramp loading. Body
cross section was generally oval with flattened top, bottom,
and sides in the vicinity of the wing. Principal dimensions
of the fuselage are as follows:

Length	79.88 in.
Maximum width	13.94 in.
Maximum depth	14.85 in.

Locations of the wing and empennage are also illustrated in
Figure 15. The wing was essentially buried in the down posi-
tion, protruding only slightly above the fuselage crown line.
At the rear of the wing center section over the fuselage, a
spring loaded fairing was located. This fairing, hinged at
its aft end to the top of the fuselage, was designed to slide
along the top surface of the wing as the wing was tilted.

With the wing down, zero wing tilt, the fuselage/wing junctures
were smoothly contoured. A fuselage cut-out was provided at
the leading edge of the wing to enable the slat to extend over
the wing center section as the wing was tilted.

Figure 15 shows the vertical and longitudinal locations of
the fuselage balance with reference to the fuselage and wing.
Note that the balance was located directly below the wing
pivot.

2.5 HORIZONTAL TAIL POSITION AND GEOMETRY

The horizontal tail was positioned high on the fin as illus-
trated in Figure 15. An additional horizontal tail position,
mid-fin, was available, however, this tail height was not
utilized during Phase I test BVWT061.

Figure 16 depicts the geometry of the horizontal tail. This
tail pivoted for stabilizer angular motion about a line per-
pendicular to the aircraft axis of symmetry and passing
through the quarter chord point of its mean aerodynamic chord
is shown in Figures 15 and 16. Stabilizer angles from +45°
to -15° were available in 5° increments. Note in Figure 15,
that with the maximum stabilizer angle of +45°, the leading
edge of the root "unported" slightly.

Primary geometry characteristics of the horizontal tail are
listed below.

Tail area, S_H 2.764 ft.^{2}

29

Taper ratio	0.609
Aspect ratio	4.646
Tail arm, ℓ_H	4.288ft.
Tail volume coefficient, $V_H = \dfrac{S_H \ell_H}{Sc}$	1.330
Dihedral	0°

2.6 VERTICAL TAIL LOCATION AND GEOMETRY

The location and geometry of the vertical tail used on Model VRO68Q is shown in Figure 17. This fin, which utilized a NACA 0012 airfoil section, was positioned on the aft fuselage as illustrated in the noted figure and was swept back 40.4° (c/4 line) for the purpose of maximizing the tail arm of the horizontal tail mounted in a high position on the fin. Geometrical characteristics of interest follow.

Tail area, S_V	1.885 ft^2
Taper ratio	.518
Aspect ratio	1.271
Tail arm, ℓ_V	3.528 ft
Tail volume coefficient, $V_V = \dfrac{S_V \ell_V}{Sb}$	.0832

30

B.L. 0

B.L. 6.91
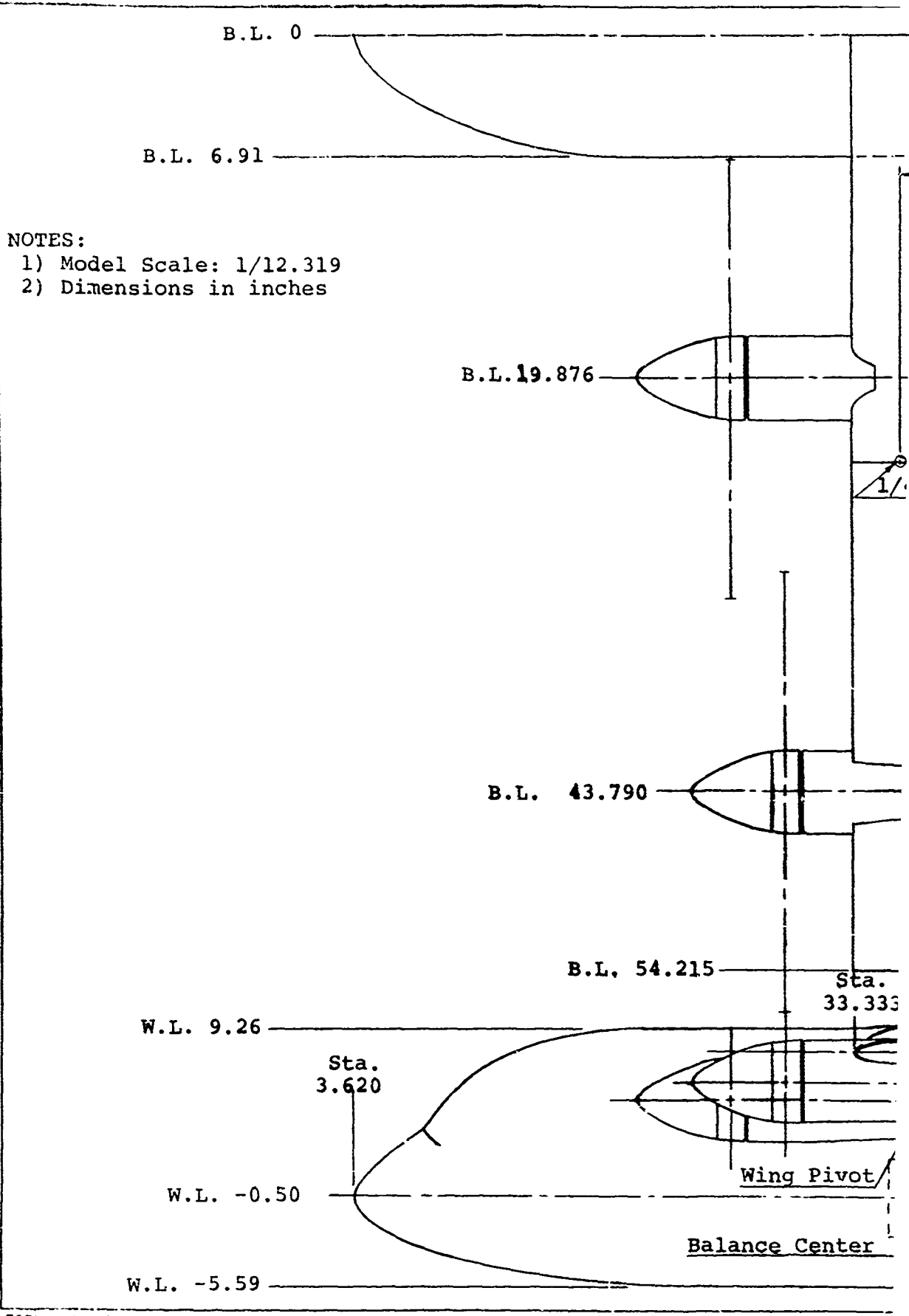
NOTES:
1) Model Scale: 1/12.319
2) Dimensions in inches

B.L. 19.876

B.L. 43.790

B.L. 54.215

Sta.
33.333

W.L. 9.26

Sta.
3.620

W.L. -0.50

Wing Pivot

Balance Center

W.L. -5.59

Figure 15

51.458"

MAC

1/4c

Hinge Line

B.L. 21.50
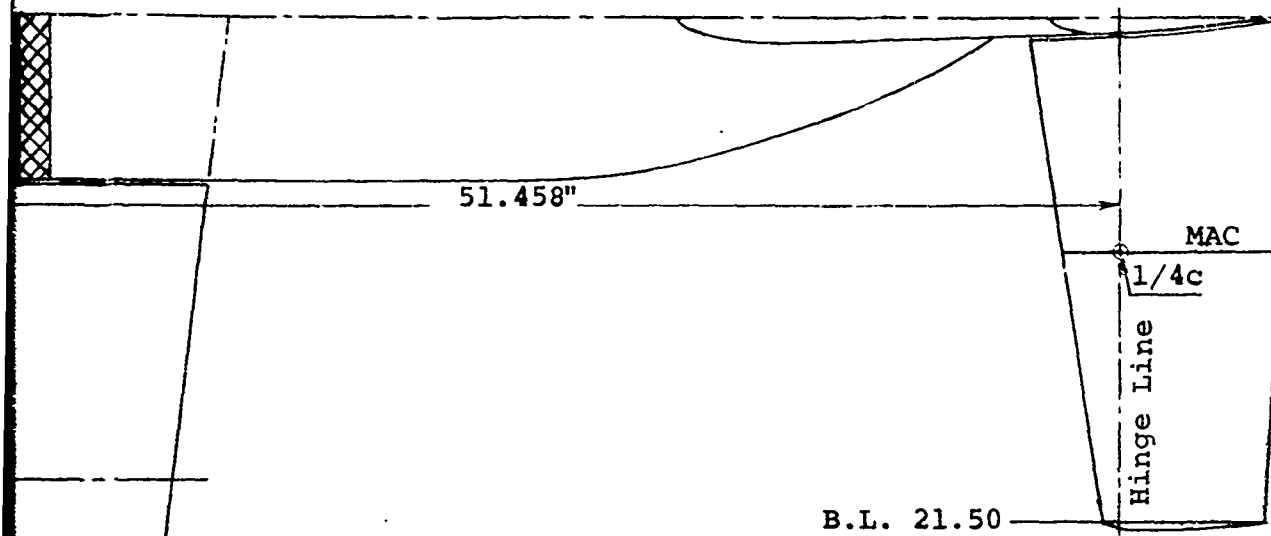
GENERAL ARRANGEMENT OF
FULL SPAN MODEL VRO68Q

MAC

Sta.
78.687

W.L. 23.490
-15°
W.L. 23.035

45°

MAC

1/4c

Sta.
66.708

Sta.
38.476

W.L. 7.812
W.L. 6.402

W.L. 6.91
W.L. 5.30

Sta.
83.50

W.L. -0.50

Model Sting Extension

MODEL VRO69Q

HORIZONTAL TAIL GEOMETRY
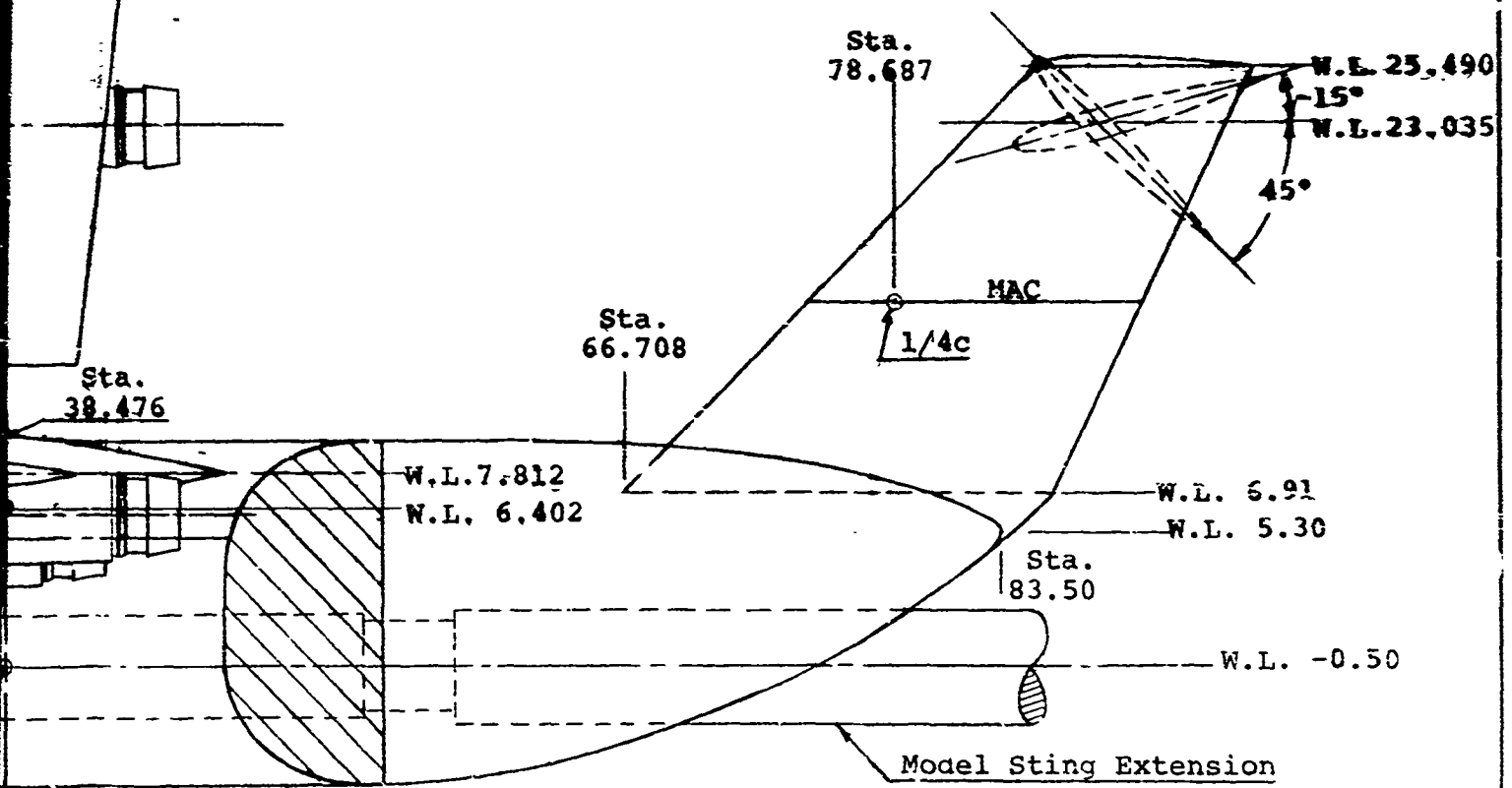
NOTES:

1) No anhedral or dihedral

8.5°

43.00"

NACA 0015

11.51"

Fin/Body ₵

Stabilizer Pivot Center

FWD

3.85"

Leading Edge

c/4

MAC

NACA 0012

7.01"

Figure 17

MODEL VR068Q VERTICAL TAIL GEOMETRY
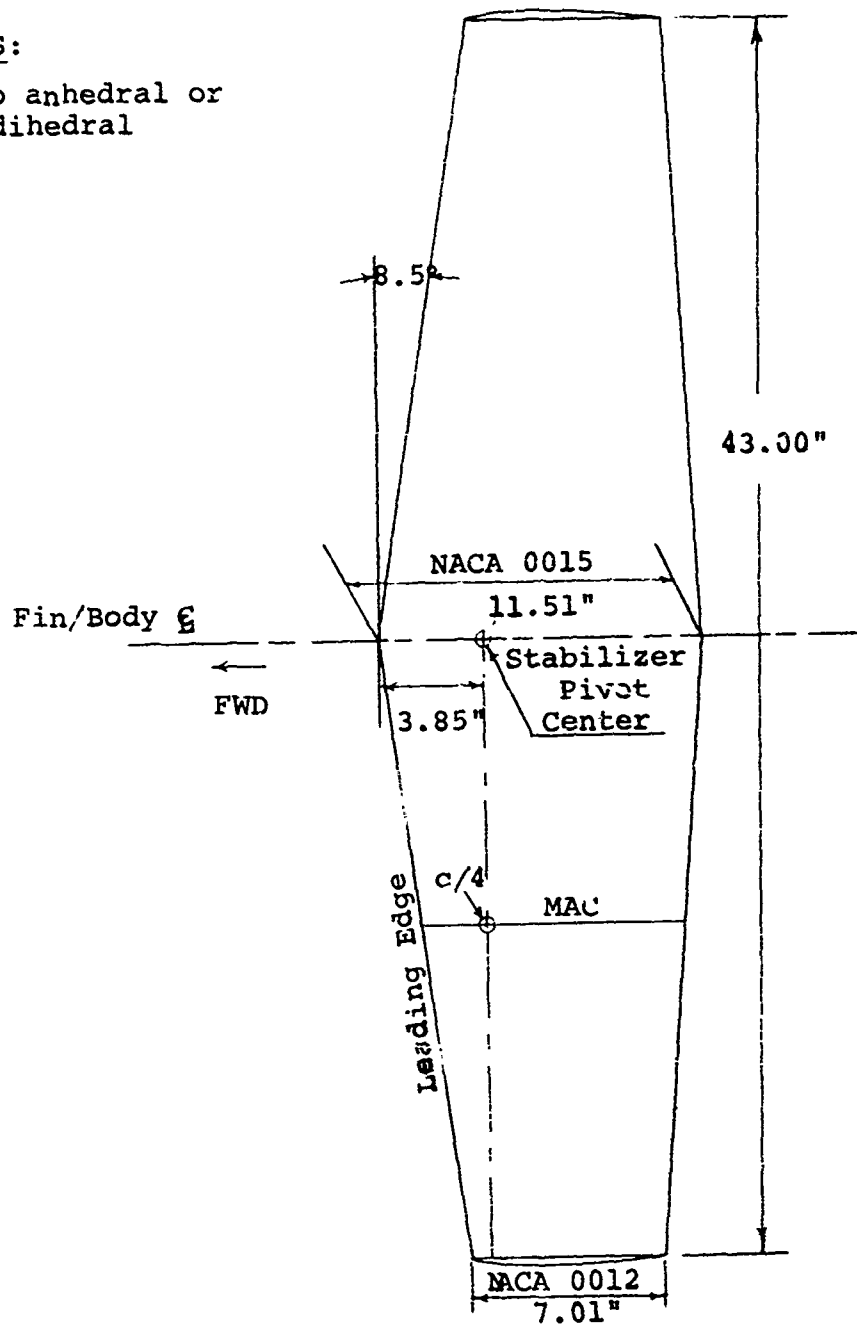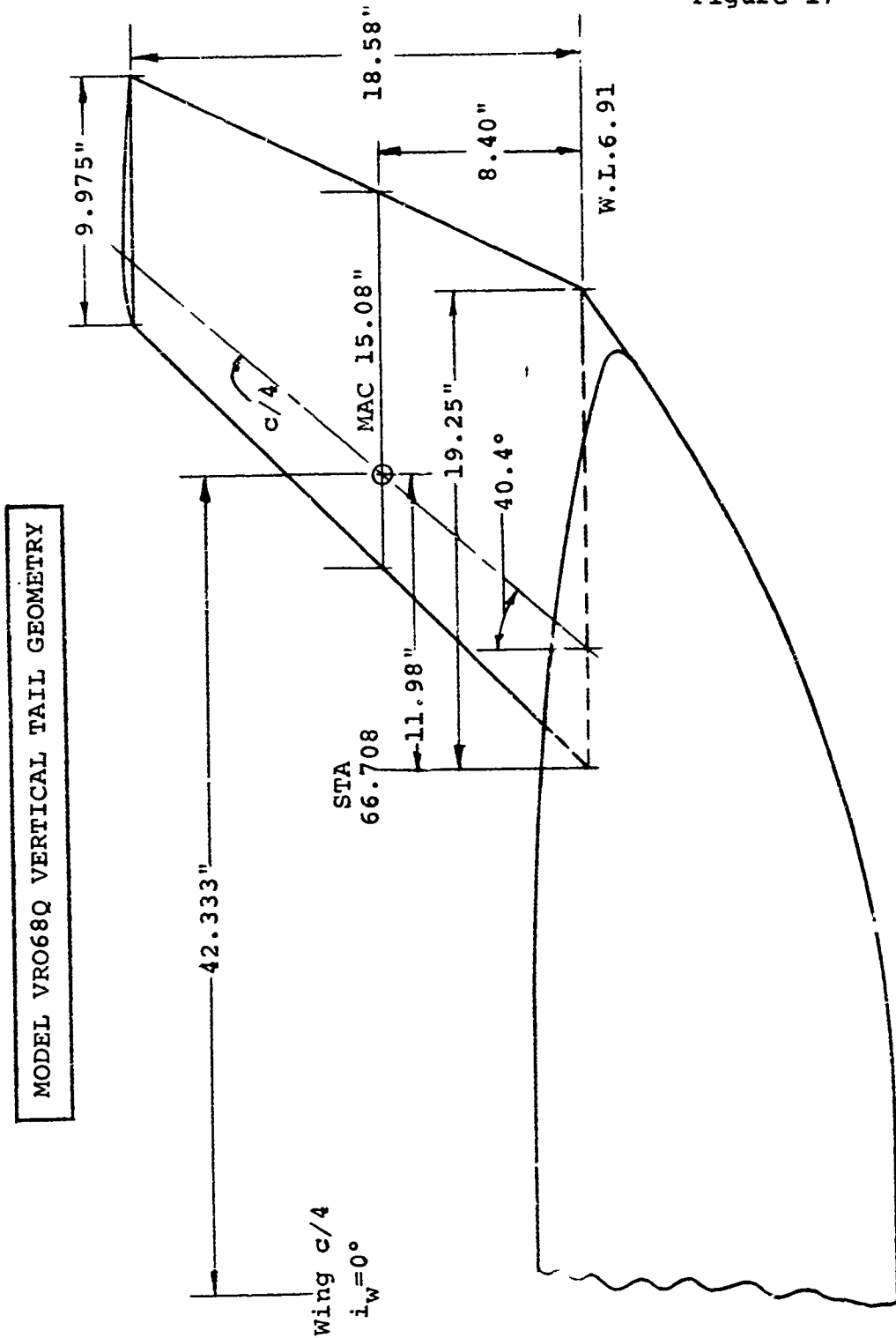
2.7 MODEL INSTALLATION

A schematic drawing of the model air supply system for powering the four pneumatic motors of Model VRO68Q is shown in Figure 18. High pressure air enters the model through the hollow sting extension. Interactions of the model air supply system on the fuselage balance measurements were minimized by ducting the air symmetrically past the balance from the forward section of the sting via dual ducts (one per fuselage side) and thence into a plenum chamber located forward of the balance in the frontal portion of the fuselage. A set of internal flexible bellows were used to connect the dual ducts to the plenum chamber structure.

Air for each air motor was individually ducted forward from the front wall of the plenum chamber, through separate motor control valves, and then aft over the top of the plenum chamber via four pipes which were connected to a hollow segmented air pivot joint. Four internal wing spanwise air ducts (one per motor) were used to direct the air outboard from the wing pivot joint into the forward portion of the air motors bolted directly to the wing.

Mass flow into each motor was remotely controlled by the four individual motor control valves used in conjunction with the main tunnel compressor system controls which established the plenum chamber pressure.

Model VRO68Q utilized the main tunnel hydraulically driven sting support system. The 16 ft. long sting pivots, for model angle of attack motion, about its attachment point on a vertical moving strut, which enables the model to be retained near the center of the test section as the model is pitched. A "yaw adapter" that provides pure yawing motion for selected angles of attack, was attached to the forward end of the main sting. This "yaw adapter" also incorporates a horizontal pivot and pin arrangement for manually setting the desired "pre-bend" angle between the fuselage centerline of the model and the centerline of the main sting.

The desired wing angle of attack range for a prescribed combination of wing tilt angle and thrust coefficient was achieved by selecting the proper "pre-bend" angle. With zero "pre-bend", the available fuselage angle of attack range is -20° to +12°. The -20° angle is the limit imposed by the maximum up-travel of the vertical strut (contact with the tunnel ceiling). The maximum positive angle of the sting with respect to the tunnel centerline (+12°) results from the limit imposed by the minimum bend radius of the 3 inch diameter (I.D.) braided steel model air hose passing up the vertical strut, through the main sting and "yaw adapter", and into the tilt wing model sting extension which is bolted to the forward end of the yaw adapter.

34

Selection of a positive 10° "pre-bend" angle provides a
fuselage angle of attack range of -10° to +22°, for example.

Figure 19 illustrates the installation of Model VRO68Q on the
main sting/strut assembly in the Boeing-Vertol V/STOL wind
tunnel. Noted on the sketch are the locations of the pitch
rotational center on the vertically moving strut and the yaw
rotational center on the yaw adapter.

Plenum

NOTES:
1) Pressure to plenum remotely
   controlled by main tunnel
   air valve

2) Mass air flow to each air motor
   remotely controlled by individual
   control valves in fuselage

Internal Flexible
Bellows (one per side)

Motor Control
Valves

Plenum

3c - (c)
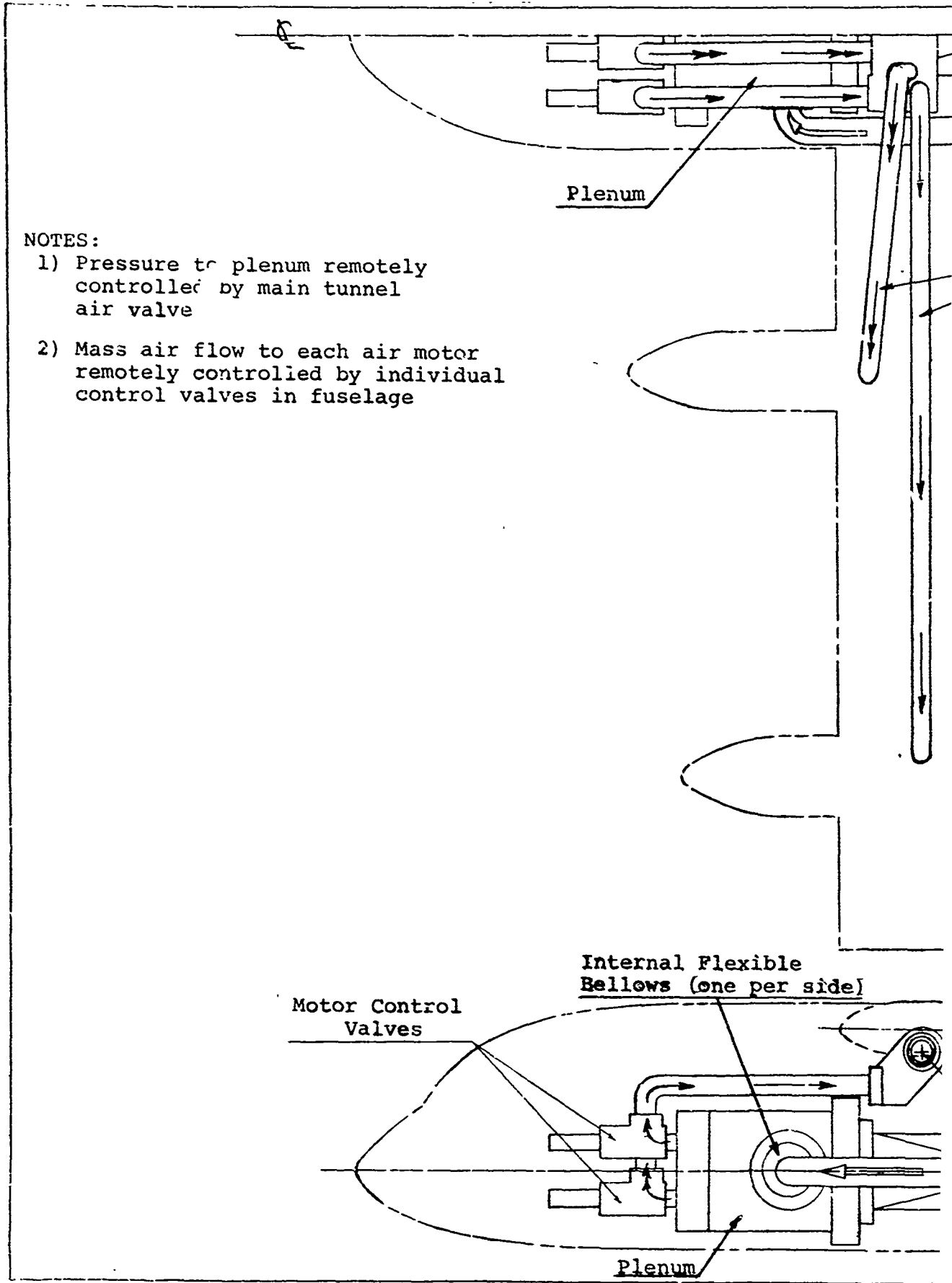
Figure 18

Internal Wing
Air Ducts

MODEL . RO68Q
SCHEMATIC DRAWING OF
MODEL AIR SUPPLY SYSTEM

Hollow Wing
Pivot

Balance
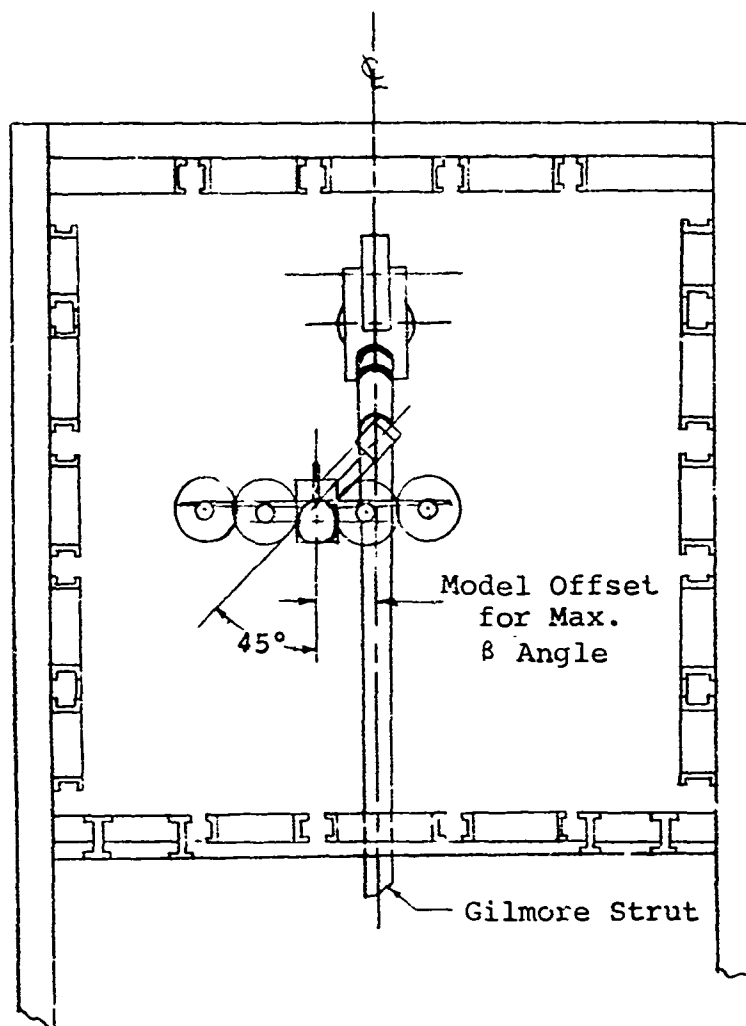
Sealed Slots
(2) per Side Wall

Model Offset
for Max.
β Angle

45°

Gilmore Strut

Figure 19

Sta. 1000

Pitch Rotational Center

Prebend
20° Angle

Balance Center

Yaw Rotational Axis

For hover phase, walls and roof downstream
of this station removed, floor slots sealed.
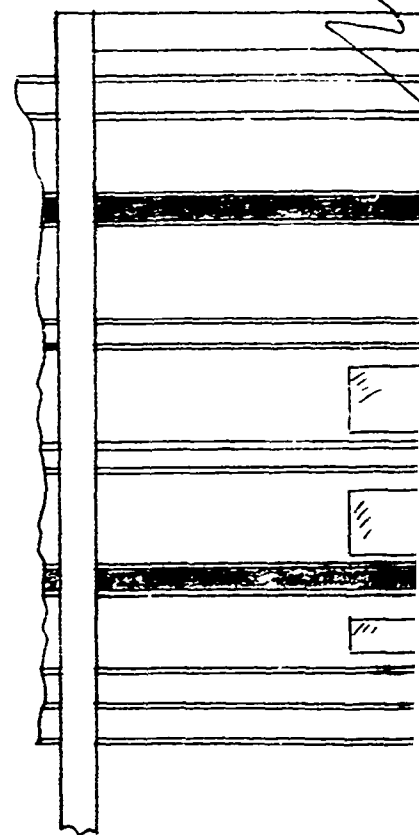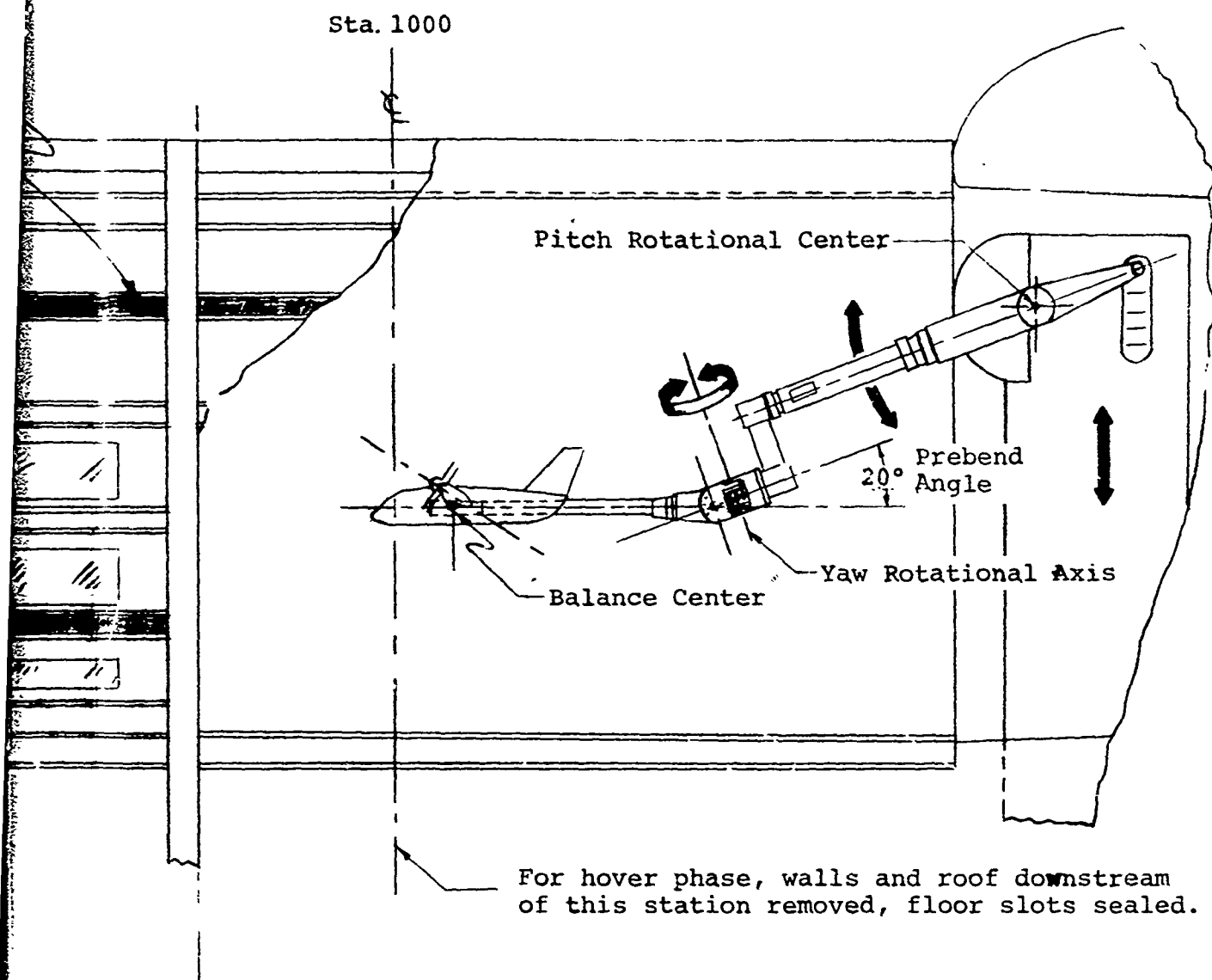
INSTALLATION OF MODEL VRC68Q
IN BOFING-VERTOL V/STOL WIND TUNNEL

2.8 TEST FACILITY

As mentioned previously, the test was conducted in a 20 ft
high x 20 ft wide x 29 ft long test section of the Boeing-
Vertol V/STOL wind tunnel. A schematic view of the facility
is presented in Figure 20.

The slotted throat test section configuration was used for the
transition or forward flight portion of the test. This tunnel
configuration is obtained by removing covers from slots built
into the test section walls, floor and ceiling. The slotted
floor was replaced with the moving belt ground plane assembly
for the in ground effect testing.

During the hover phase of the test, the 29 ft long x 20 ft high
test section walls were removed and lowered into pits. The
raising of the test section ceiling to the top of the 67 ft
diameter plenum chamber that surrounds the test section pro-
vided a hover test area with a height, as measured from the
solid test section floor used as a ground plane, of approxi-
mately 50 ft.

The auxiliary air for powering the four nacelle pneumatic
motors was supplied by a 20 pound per second, 1000 psi com-
pressor system.
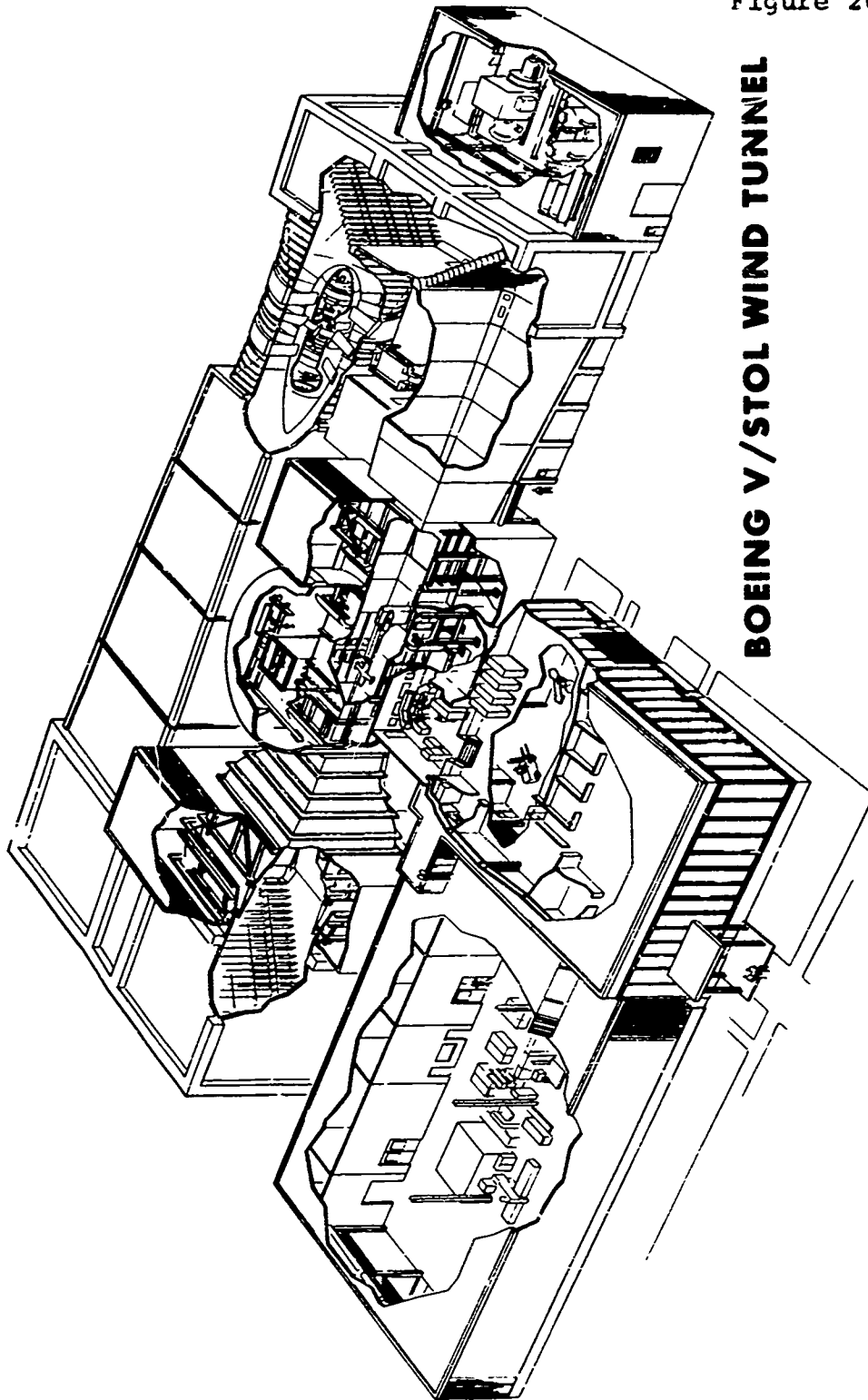
BOEING V/STOL WIND TUNNEL

3.0 INSTRUMENTATION AND EQUIPMENT

3.1 MODEL INSTRUMENTATION

Model instrumentation consisted of the following items:

a) Six Component Strain Gage Fuselage Balance

Total aircraft normal force, axial force, pitching moment,
yawing moment, rolling moment, and side force were measured
by this balance, that was located with its longitudinal
axis parallel to the body centerline and with the balance
center directly below the wing pivot as shown in Figure 15.

b) Six Component Strain Gage Balance in Each Nacelle

Identical balances were used in the outboard and inboard
nacelles and were similarly located as shown in Figures 13
and 14. Each balance measured propeller thrust, pitching
moment, normal force, yawing moment, side force, and roll-
ing moment. The rolling moment in this case was the fric-
tion torque produced by the bearings and cyclic pitch me-
chanism. In addition to measuring steady values, normal
force, pitching moment, and propeller thrust from selected
nacelles were displayed on oscilloscopes so that dynamic
loads in the balance flexures could be monitored to pre-
vent exceeding the fatigue allowables.

c) Strain Gaged Propeller Drive Shafts

Each nacelle assembly incorporated a high speed slip ring
assembly for the purpose of transmitting electrical signals
from the strain gage bridges measuring torque.

d) Tachometers

The rotational speed of each propeller was measured by a
tachometer, installed internally behind the front bearing
of the air motor. This type of tachometer worked on the
pulse generator principle (thirty pulses per cycle). Four
digital voltmeters were used for direct readouts following
amplification of the signals.

e) Wing Tilt Angle Potentiometer

Wing tilt angle was measured using a precision potentio-
meter. A gearing arrangement was used to prevent potentio-
meter slippage due to model motions.

f) Bearing Temperature Thermocouples

Each propeller shaft thrust bearing, located in the nacelle

40

stack, was instrumented. The four bearing temperatures were displayed on the control panel for model condition monitoring purposes.

g) <u>Air Passage Metering Valve Position Indication</u>

Mass flow into each air motor was individually controlled by a metering valve. The position of these four valves was displayed on the overhead control panel as an aid in setting up the four air motors for a constant RPM operation.

h) <u>Nacelle Balance Rolling Moment (Friction Torque)</u>

The rolling moment signal from each of the nacelle balances was displayed in the overhead panel on one of the safety meters used for monitoring dynamic data. Any increase above the characteristically low friction torque in a cyclic hub/air motor assembly would be immediately recorded by the nacelle roll balance, permitting a rapid shut down if necessary.

3.2 <u>DATA ACQUISITION SYSTEM</u>

The flow diagram of the wind tunnel data system used in this test is shown in Figure 21. This data system can accept up to 120 channels from the model and the tunnel itself. These signals are routed as illustrated to an IBM 1800 computer for processing and data reduction. The computed results are tabulated by a line printer and selected quantities are plotted by the X-Y plotters. Final data is stored on magnetic tape.

A digital display of any nine channels is also available during testing for monitoring purposes. Dynamic data of six quantities can be continuously displayed on oscilloscopes. This provides assistance in preventing balance or structural limits from being exceeded.

A choice of sampling rates in terms of samples per channel/ sec. and in sampling time periods is available. The sampling process is accomplished with channel switching devices called multiplexers.

TUNNEL SENSORS

MODEL SENSORS

SAFETY METERS

SIGNAL CONDITIONING

OSCILLOGRAMS

DYNAMIC MULTIPLEXER AND ADC*

QUASISTATIC MULTIPLEXER AND ADC*

DIGITAL DISPLAY

IBM 1800 DATA PROCESSOR

X-Y PLOTS

DISK STORAGE

LONG LINE DATA ADAPTER

LINE PRINTER

*Analog to Digital Converter

MAGNETIC TAPE STORE

CARD READ/PUNCH

INPUT/OUTPUT TYPEWRITER

DATA ACQUISITION SYSTEM

42
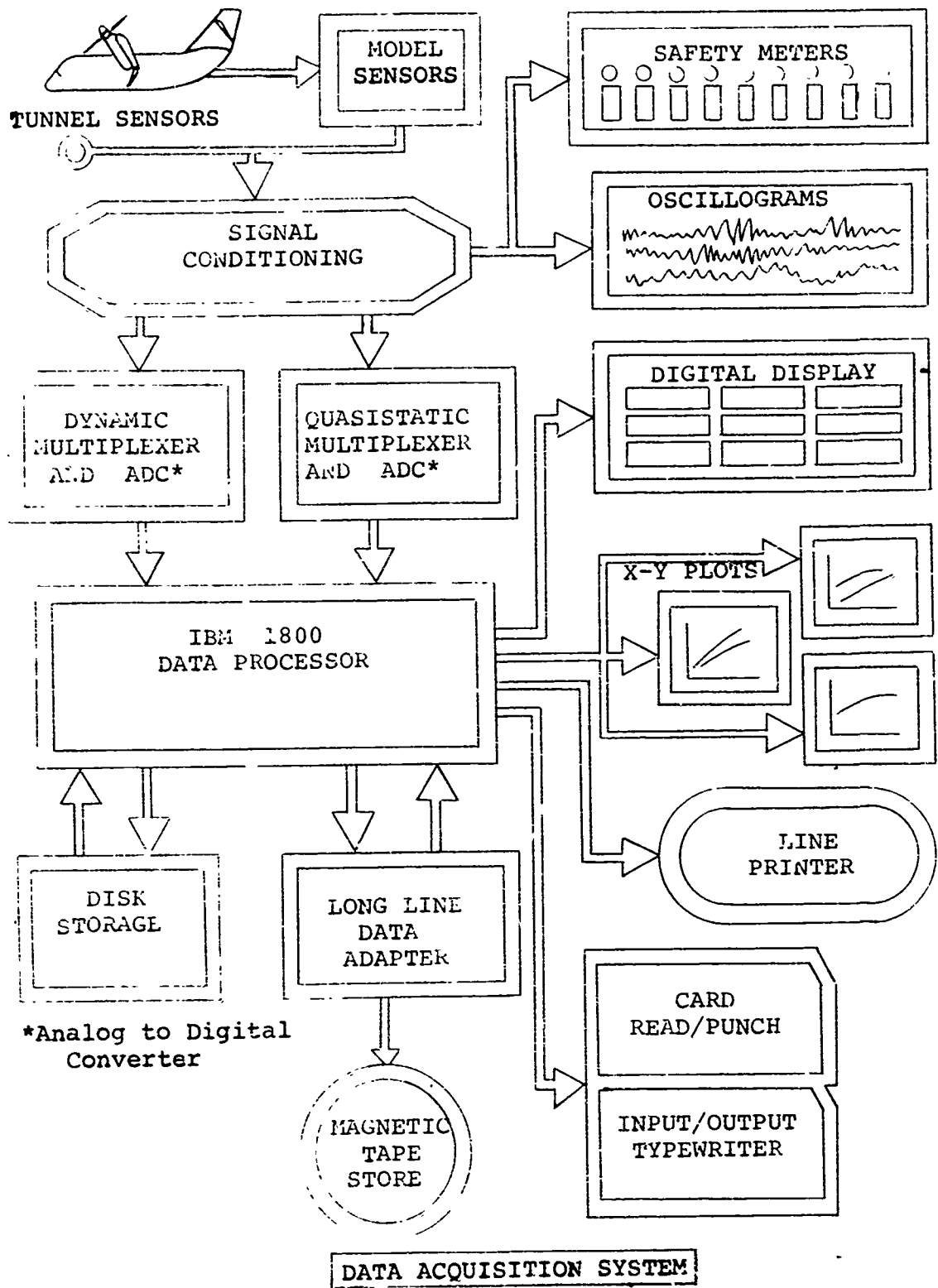
4.0 DATA REDUCTION

At each test point, measurements were taken for computing and printing out on-line the quantities listed below.

Air supply line pressure	psi
Density, ρ	slugs/ft^3
Freestream dynamic pressure, q	lb/ft^2
Fuselage angle of attack, α_F (correction applied for sting deflection)	degrees
Sideslip angle, β	degrees
Model height, h	inches
Propeller speed (each propeller)	RPM
Shaft torque (each propeller)	ft.lbs.
Tunnel velocity, V	ft/sec
Wing tilt angle, i_w	degrees

The following aircraft forces and moments were measured by the fuselage balance.

Normal force (positive: up)	lbs.
Axial force (positive: forward)	lbs.
Pitching moment (positive: nose up)	ft.lbs.
Yawing moment (positive: nose right)	ft.lbs.
Rolling moment (positive: right wing down)	ft.lbs.
Side force (positive: to the right)	lbs.

Balance interaction corrections to these measured forces and moments were calculated and applied on-line using the balance interaction matrix incorporated into the data program. Static pressure tares, resulting from the high pressure mod.. ir supply lines extending forward from the model sting ension into the plenum chamber located in the frontal portion of the fuselage, were also applied on-line. The static pressure tare curves inserted into the data program were linear and of the following magnitude at 220 psi,a typical operating line pressure in hover.

43

Normal force: -2.2 lbs.

Axial force: -4.5 lbs.

Pitching moment: 0 ft.lbs.

Yawing moment: +1.2 ft.lbs.

Rolling moment: 1.6 ft.lbs.

Side force: -1.4 lbs.

At each test point, the value of jet thrust produced by each
air motor was established by entering the jet thrust correction
"look-up" tables incorporated into the computer program as a
function of shaft torque. The components of jet thrust in-
herent in the measured values of aircraft normal force, axial
force, and pitching moment were extracted on-line, as a func-
tion of wing tilt angle.

Finally, model weight tare values for a particular wing tilt/
fuselage angle combination and model configuration were deter-
mined and applied on-line using the appropriate weight tare
equations inserted into the data program.

Aircraft normal force and axial force from the fuselage bal-
ance were resolved on-line into the wind axis system in order
to compute

Lift, L   lbs.

Longitudinal force, X(positive forward)   lbs.

Aircraft pitching moment measured by the fuselage balance was
transferred on-line to the wing pivot for runs wherein wing
tilt angle was varied (fuselage angle was held constant) and
to an aircraft c.g. position representative of the wing tilt
angle for runs wherein fuselage angle was varied (wing tilt
angle held constant)or yaw angle was varied.

Aircraft forces and moments were reduced on-line to the follow-
ing coefficient form based on slipstream dynamic pressure.

Lift coefficient, $C_{L_s} = \dfrac{L}{q_s S}$

Longitudinal force coefficient, $C_{X_s} = \dfrac{X}{q_s S}$

Pitching moment coefficient, $C_{M_s} = \dfrac{M}{q_s c S}$

(where c=MAC of tapered wing)

Side force coefficient, $C_{y_s} = \dfrac{\text{Side force}}{q_s S}$

Yawing moment coefficient, $C_{n_s} = \dfrac{\text{Yawing moment}}{q_s S b}$

Rolling moment coefficient, $C_{\ell_s} = \dfrac{\text{Rolling moment}}{q_s S b}$

Aircraft lift, longitudinal force and pitching moment were also reduced on-line into the following non-dimensional form.

$$\dfrac{L}{qb^2} \quad , \quad \dfrac{D}{qb^2} \text{ (positive aft for drag), } \dfrac{M}{qb^2 c}$$

Each of the four internal nacelle balances measured the following forces and moments.

Thrust, T (positive:forward)	lbs.
Prop normal force (positive:up)	lbs.
Prop pitching moment (positive:nose up)	ft.lbs.
Prop side force (positive:to the right)	lbs.
Prop yawing moment (positive: nose right)	ft.lbs.
Prop rolling moment (same as friction torque)	ft.lbs.

A balance interaction matrix for each nacelle balance, as developed from the static calibration, was incorporated into the on-line computer program. Interaction corrections to the measured propeller forces and moments were calculated and applied using the appropriate nacelle interaction matrix. Weight tares due to the weight of the propeller/hub assembly were then calculated and corrections applied on-line.

Pitching moment and yawing moment that were measured about the balance center were transferred to the hub $\underset{\text{L}}{\mathbb{C}}$ so that hub moment coefficients could be calculated at the plane of the propeller. The following propeller-type coefficients were computed and printed out on-line for each of the four propellers.

Advance ratio, $J = \dfrac{V}{nD}$

Thrust coefficient, $C_T = \dfrac{T}{\rho n^2 D^4}$

Prop pitching moment coefficient,
$$C_{M_p} = \dfrac{\text{Prop pitching moment}}{\rho n^2 D^5}$$

Prop shaft power coefficient, $C_p = \dfrac{\text{Shaft power}}{\rho n^3 D^5}$

$$= \dfrac{2\pi (\text{Shaft torque})}{\rho n^2 D^5}$$

Prop normal force coefficient, $C_{NF_p} = \dfrac{\text{Prop normal force}}{\rho n^2 D^4}$

Prop side force coefficient, $C_{SF_p} = \dfrac{\text{Prop side force}}{\rho n^2 D^4}$

Prop yawing moment coefficient, $C_{YM_p} = \dfrac{\text{Prop yawing moment}}{\rho n^2 D^5}$

Friction power coefficient,

$$C_{P_F} = \dfrac{2\pi (\text{Balance rolling moment})}{\rho n^2 D^5}$$

Coefficients of prop thrust, prop pitching moment, and prop normal force in slipstream notation were also calculated and printed out on-line for each propeller per the following listing.

Thrust coefficient, $C_{T_S} = \dfrac{T}{q_s A_p}$

Prop pitching moment coefficient, $C_{MP_S} = \dfrac{\text{Prop pitching moment}}{q_s S c}$

Prop normal force coefficient, $C_{NF_S} = \dfrac{\text{Prop normal force}}{q_s S}$

where $q_s = q + T/A_p$

Full scale aircraft rate of descent in ft/min and velocity in knots were computed on-line at each test point by the following equations inserted in the data program.

$$R/D = 60\left(\dfrac{D}{qb^2}\right) \cdot \sqrt{\dfrac{2W}{\rho b^2}} \cdot \sqrt{\dfrac{1}{\left[\left(\dfrac{L}{qb^2}\right)^2 + \left(\dfrac{D}{qb^2}\right)^2\right]^{3/2}}}$$

$$V_F = .592\sqrt{\left(\dfrac{2W}{\rho b^2}\right)\left(\dfrac{qb^2}{L}\right)}$$

46

The weight term used in the above equations was commensurate
with a typical full scale tilt wing aircraft operating at its
maximum gross weight in the "V" mode which corresponded to a
wing loading (W/S) of 66 lb/ft^2. This value was increased to
73.5lb/ft^2 to account for the difference between the average
tunnel density used in the above equations (ρ=.00228 slugs/ft^3)
and the typical atmospheric design conditions of 3000ft/90°F
(ρ=.00204 slugs/ft^3) for a full scale tilt wing.

The following equation was used in the data program to compute
propeller induced velocity, w, in ft/sec.

$$w^4 + w^3\ 2V \cos i_w + w^2 V^2 = \left(\frac{T}{2\rho A_p}\right)^2$$

The above parameter was used to determine the effective wing
angle of attack, $\alpha_{W_{EFF}}$, at buffet onset. This angle is defined
as the angle between the wing chord and the resultant velocity
at the wing. The resultant velocity was obtained by adding
vectorially, the tunnel velocity and the **induced** velocity
at the leading edge of the wing, assuming full contraction of
the propeller wake at the wing leading edge.

Three wind tunnel X-Y plotters were used to produce on-line
plots of the following during pitch sweeps.

Force polars in terms of C_{L_S} vs C_{X_S}

or L/qb^2 vs D/qb^2

Lift curves in terms of C_{L_S} vs α_f

or L/qb^2 vs i_w

Pitching moment curves in terms of M/qb^2c vs i_w

or C_{M_S} vs α_F

During yaw sweeps these plotters were used to develop the
following on-line plots.

Yawing moment curves in terms of C_{n_S} vs β

Rolling moment curves in terms of C_{ℓ_S} vs β

Side force curves in terms of C_{Y_S} vs β

No tunnel wall corrections were applied to the data.

47

5.0   TEST PROCEDURE AND TEST CONDITIONS

5.1   TEST PROCEDURE

The fuselage and four nacelle strain gage balances were check
calibrated statically with the model built-up and installed
in the test section (i.e., plumbing was installed and the
nacelles were assembled) by applying known forces and moments
on each balance. These applied forces and moments were com-
pared to those calculated and printed out on-line, via the
computer, using the prime balance sensitivities developed
during the previous Phase I test of the same model. The re-
sultant calibration checks confirmed the established balance
calibrations and showed that sensitivity adjustments were re-
quired in only a couple channels of the four nacelle balances.
Balance interaction matrices utilized in the on-line computer
program where those developed for the Phase I test and included
in the case of the fuselage balance interaction matrix, the
constraining effect of the plumbing for the air motors.

The installed model was then statically pressurized in incre-
ments to determine the effect on the measured forces and mo-
ments from the fuselage balance, of the high pressure air
lines extending forward from the sting, across the fuselage
balance, and into the plenum chamber located in the nose of
the fuselage. Static pressure tares developed from this cali-
bration were incorporated into a look-up routine in the data
program so that corrections could be applied on-line to forces
and moments sensed by the fuselage balance.

Sting deflections due to the net vertical force (lift minus
model weight) acting on the model were established by hanging
known weights on the model and measuring the true fuselage
angle of attack with an inclinometer. The resultant correc-
tion curve, showing incremental fuselage angle of attack as a
function of net vertical force, was inserted into the computer
program so that the indicated fuselage and wing angle of
attack could be adjusted on-line to true values.

Due to the different resultant stiffnesses of the inboard and
outboard nacelle when mounted on the model, it was necessary
to dynamically balance each of the propeller/hub assemblies
in place on the wing. The balancing target in each case was
+10% of the allowable fatigue load for nacelle balance pitch-
ing moment and normal force, which were monitored via oscillo-
scope displays during the test.

The jet thrust produced by each air motor was determined by
individually running each motor in place on the model with a
cylindrical cross section "propeller" installed and zero

tunnel q. This tubular "propeller" was utilized so that the air motor torque could be absorbed without the simultaneous production of propeller thrust. The calibration curves as obtained from resolving fuselage balance yawing moment are presented in Figure 22, as a function of shaft torque. These curves, which are in agreement with the corresponding calibration curves developed during the Phase I test (see Figure 20 of Reference 1 test report), were incorporated into the computer program so that the jet thrust produced by each motor could be extracted on-line as a function of wing tilt angle from the forces and moments measured by the fuselage balance.

Of interest is the variation of shaft torque with RPM during the jet thrust calibration runs, the data for which is shown in Figure 23. Since the cylindrical tube prop operates in the static condition at basically a constant torque coefficient (Reynolds number effects on drag coefficient excluded), the variation of shaft torque should be approximately linear with RPM^2. In addition, as a result of utilizing the same tubular prop for the various calibration runs, the same shaft torque should have been recorded at a particular RPM on each of the calibrations runs. The data shown on Figure 23 is in general agreement with these comments, however, a comparison of this figure with the similar Figure 169 presented in Appendix B of the Phase I test report will show that at 5000 RPM, a typical propeller operating speed, the shaft torque measured for the the Phase II test was 8.5% greater. Of this 8.5% value, 5% is accounted for by the higher air density prevailing during the Phase II air motor calibration runs and the remaining 3.5% is accounted for by the higher calibration sensitivities obtained during the Phase II test when the instrumented shafts were calibrated in place on the assembled model with the aid of a special calibration fixture used to lock the air motor.

Two separate sets of propeller hub assemblies (collective hubs and cyclic pitch hubs that were designed for the 4500-5000 RPM operating regime) are available for Model VRO68Q, both of these being used during the Phase I test program. Since the majority of the data runs planned for the Phase II test program were directly involved with investigating cyclic pitch inputs, the primary exception being the testing devoted to establishing the lateral/directional stability characteristics in transition, a decision was made to conduct the entire Phase II test with the cyclic pitch hub assemblies. A decision otherwise would have necessitated the dynamic balancing plus thrust balancing of two separate hub assemblies and interchanging the hub systems several times during the program to match the different modes of testing.

49

Prior to performing the actual data runs with the propeller hub assemblies installed, it was necessary to balance the thrusts from the four propellers. This procedure was straightforward in the initial hover portion of the test, wherein the collective blade angle of each propeller was manually adjusted until the maximum thrust difference between propellers was within 1% of the total thrust from the four propellers. A total propeller thrust of 241 lbs. was developed with 5000 RPM and an average blade angle of 12.5° for the out of ground effect hover condition. This thrust level compares to a Phase I test value of 230 lbs. for the same condition. During the forward flight portion of the test, it was necessary to examine the thrust balance at the tunnel dynamic pressures and representative wing tilt angles for which data was to be acquired and adjust the collective settings accordingly.

Weight tares were taken for each significant model configuration. Subsequent to the incorporation of the weight tare data into the on-line data program, wind-off data points were acquired at a couple of model angles to ensure that the weight tare routine was accurately functioning.

As mentioned previously, test BVWT 067 was performed using three different tunnel test section configurations: (1) walls/ceiling removed and solid floor for the hover portion, (2) slotted walls/ceiling/floor for the forward flight transition portion, and (3) slotted walls and ceiling/moving belt ground plane for the in ground effect testing.

In the hover runs, ground height was varied, with the wing tilt angle being held constant at 90° and fuselage angle at zero degrees. Five values of model height above the tunnel floor were preselected, with the height measured vertically from the plane of the outboard propeller. These heights corresponded to 4.0, 2.0, 1.5, 1.25, and 1.0 propeller diameters.

A dynamic problem has been experienced on sting mounted Model VRO68Q operating in the hover mode with the wing tilted at 90°, that resulted in a scatter of the yawing and pitching moment data. The distance from the center of the fuselage balance to the tunnel sting pivot on the vertical strut was over 25 ft. This problem, which manifested itself in a long period random-type model oscillation (approximately 10 second period), was largely circumvented by increasing the time period for data sampling to 10 seconds and averaging the moments from a minimum of three data points at the most dynamically critical test conditions.

Three separate test procedures were used during the forward flight transition portion of the test: (1) wing tilt angle

pitch sweeps with the fuselage level for the descent perform-
ance runs, (2) fuselage angle pitch sweeps with a fixed wing
tilt angle for the longitudinal stability and control runs,
and (3)yaw angle sweeps with wing tilt angle and fuselage
angle fixed for the lateral/directional stability and control
runs. The model was offset laterally in the test section
with respect to the tunnel centerline by turning the "dogleg"
sting adapter 45°(See Figure 19). This offset enabled large
sideslip angles to be attained when the model was yawed to
the left. Transition runs, both longitudinal and lateral/
directional runs, were conducted with a constant tunnel q
and propeller RPM (except the zero thrust coefficient, C_{T_S},
runs, wherein propeller RPM was reduced as the model was pitched
to maintain zero propeller thrust).

For the in-ground effect testing, the slotted test section
floor was lowered, then stowed, and the moving ground belt
floor was installed in its place. The electrically driven
belt was adjusted in velocity via the belt speed control
mounted in the console. Belt speed was matched to the tunnel
speed displayed in fps on the overhead test monitor panel
(data is transmitted into the test monitor from on-line data
computations and then is transcribed into digital form).
Actual belt velocity in fps was also digitally displayed on a
direct readout counter. The electrical pulses into this
counter were generated by a magnetic pickup (320 tooth gear)
mounted on the belt follower.

In-ground effect testing was performed with a level fuselage
at a fixed ground height representative of a flight condition
with the wheels 3 ft. off the ground. Since model pitching
(fuselage angle pitching) is limited with a sting mounted
model operating in proximity to the moving ground belt and
since longitudinal characteristics were desired over a large
wing angle range, pitch data was obtained using wing tilt
angle sweeps. Lateral/directional testing was conducted using
yaw sweeps and zero prebend angle to preclude rolling the
model as the model was yawed (this would occur as a result
of the yawing axis on the sting being located aft of the pre-
bend rotational center - see Figure 19).

Maximum tunnel dynamic pressure utilized during test BVWT 067
occurred during the zero thrust runs. The lowest tunnel q
used (1.0q) was that that could be achieved with reasonable
accuracy. This q value established the maximum test value
of C_{T_S}. Tunnel dynamic pressures between these two extremes
were selected to achieve the desired spread of C_{T_S} values.

Throughout the test, dynamic loads as measured by the nacelle

51

balances (pitching moment and normal force from the four pro-
pellers and thrust of two of the propellers) were continuously
monitored via oscilloscope presentations to ensure that fatigue
loads inside the four nacelles were not exceeded. The drive
shaft bearing temperatures in each of the nacelles and nacelle
rolling moment (friction torque) were also continuously moni-
tored to prevent running with overheated or damaged bearings.

AIR MOTOR JET THRUST CALIBRATIONS

NOTES:
1. Test BVWT 067
2. Cylindrical Tube Prop
3. Vary RPM
4. Zero Tunnel q
5. Y.M. Resolution

SYM	RUN	MOTOR NO.	WING POSITION
◇	1	138	LOB
○	3	143	ROB
◻	4	139	RIB
⊡	5	139	RIB
○	6	141	LIB

53

NOTES:

1. Test BVWT 067
2. Cylindrical Tube Prop
3. Vary RPM
4. Zero Tunnel q

SYM	RUN	MOTOR NO.	WING POSITION
◇	1	138	LOB
◯	3	143	ROB
▢	4	139	RIB
▽	5	139	RIB
◯	6	141	LIB

$Q \sim$ ft-lbs

$RPM^2 \times 10^{-6}$
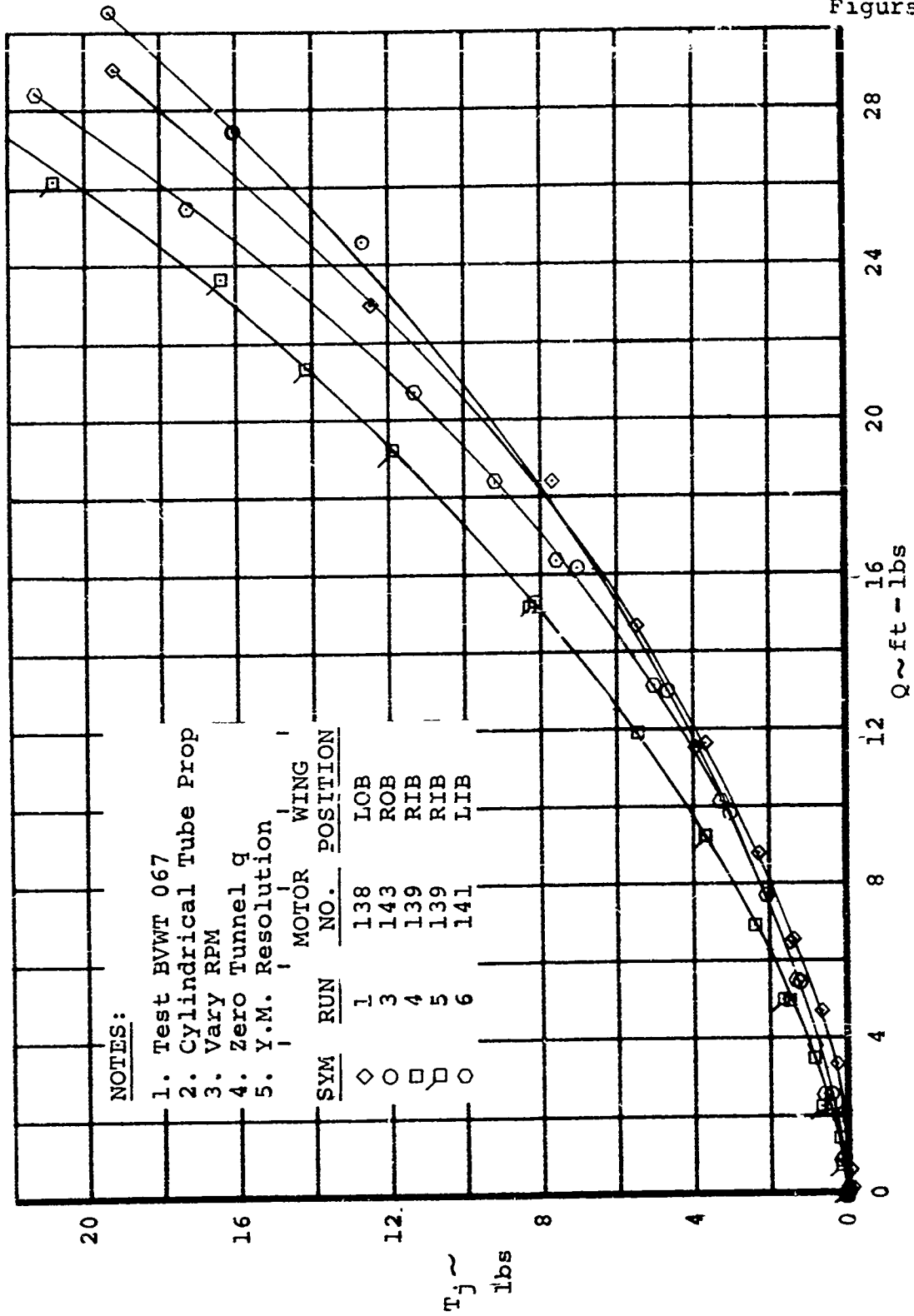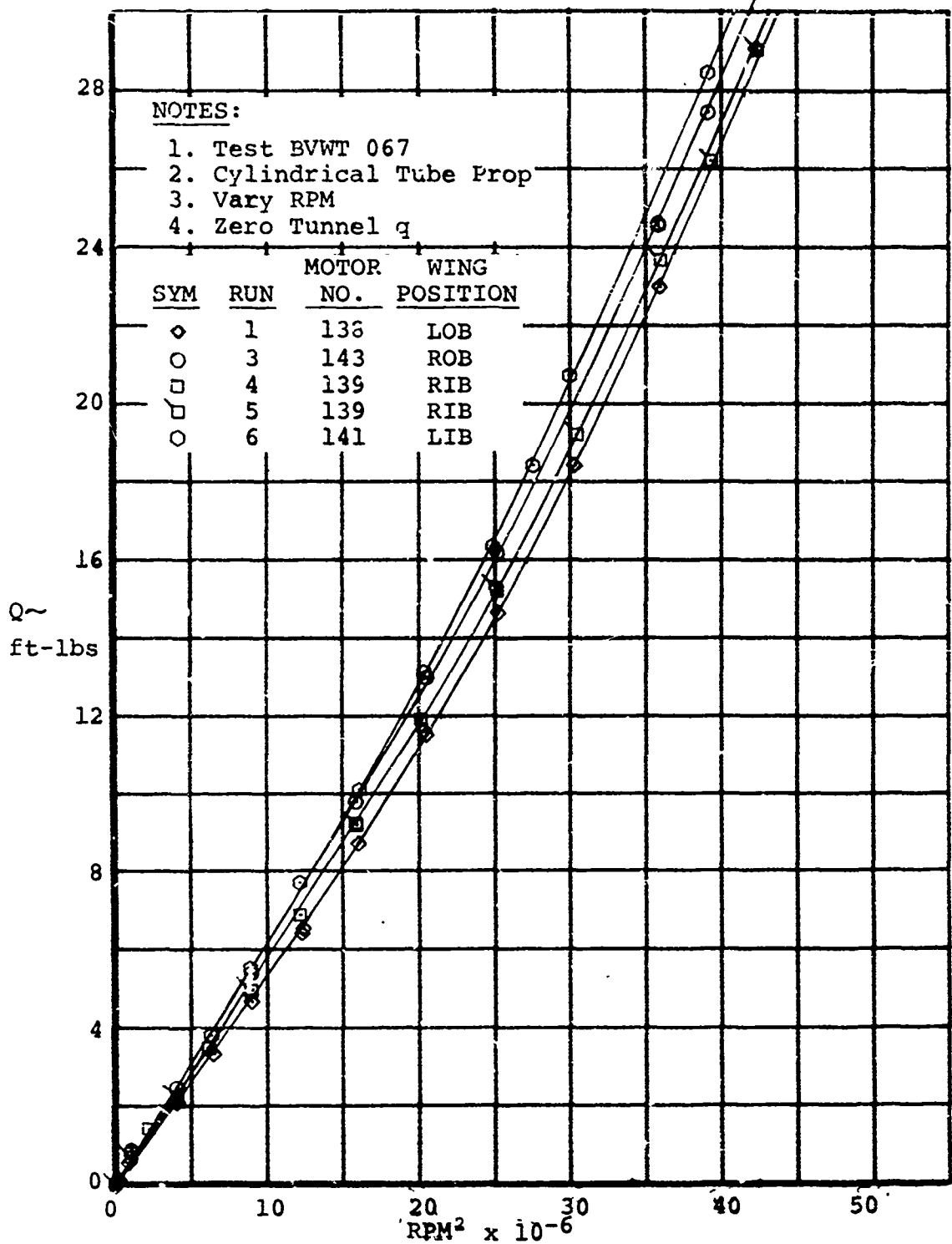
VARIATION OF SHAFT TORQUE
DURING JET THRUST CALIBRATIONS

54

5.2 TEST CONDITIONS

Tabulations on the following pages summarize the data runs
performed during the subject test (BVWT 067) in terms of key
test variables. The runs have been arranged in sets according
to test objectives. Air motor jet thrust calibrations runs
and thrust balancing runs required for "setting-up" the model,
have been deleted from the listings. All data runs were
performed with the P[1,2] prop rotation (props turning down
between the nacelles).

The range of slipstream thrust coefficient evaluated during
the forward flight longitudinal stability and control testing
with cyclic pitch varied from 0.27 C_{T_S} to a maximum C_{T_S} of 0.93.
This testing was phased so that the longitudinal stability and
trim with the smaller wing tilt angles (15°) was examined at
the low end of the C_{T_S} spectrum and large wing tilt angles
(up to 55°) were examined at the high end. Cyclic pitch in-
puts were similarly phased with the largest cyclic angles
being investigated at the highest C_{T_S} values where cyclic con-
trol requirements are a maximum. Lateral/directional stabili-
ty testing was conducted in a similar manner.

BVWT 067
RUN SUMMARY
HOVER

RUNS	CYCLIC ANGLE	FLAP ANGLE LEFT/RT WING/WING	SPOILER ANGLE LEFT/RT WING/WING	OBJECTIVE
10/11	0	0/60°	0/0	Yaw Control
12	↓	✓	60°/0	↓
13	↑	-40°/60°	↓	↓
14/15	+6°	0/60°		Effect of Cyclic on Yaw Control
16	-6°	✓	↓	✓
17	+6°	0/0	0/0	Effect of Tail on Cyclic Control

NOTES:

1. Wing tilt angle: 90°
2. Ground height varied for each run
3. Average collective setting with cyclic hubs: 12.5°
4. Slats retracted
5. Propeller speed: 5000 RPM

BVWT 067
LONGITUDINAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_s}	OBJECTIVE
21	Vary	60°	---	+4°	5000	.85	Low Speed
22			---			.75	Descent
23			---			.58	Capability
24			---	↓	↓	.37	
25			---	+6°	5000	.85	
26			---			.75	
27			---		↓	.58	
173	↓	↓	---	↓	4500	.58	↓

NOTES:

1. Wing tilt angle sweeps
2. Full span slat setting: Q*,10,10,*
3. Average collective setting with cyclic hubs: 12.8°
4. Zero fuselage angle
5. Horizontal tail off

BVWT 067
LONGITUDINAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUNS	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_S}	OBJECTIVE
31 33 39	15°	60°	5°	0 +4° -4°	5000	.30	Cyclic Pitch Control/Tail On
30 32 38	15°		5°	0 +4° -4°		.45	
50 35/61 37/70	30°		15°	0 +4° -4°		.55	
51 34/60 36/69	30°		15°	0 +4° -4°		.70	
52 54 171 65 169A	45°		35°	0 +4° +6° -4° -6°	4500 5000 4500	.81	
53 55 170 66 169	55°		35°	0 +4° +6° -4° -6°	5000 4500 5000 4500	.93	

NOTES:

1. Fuselage angle sweeps
2. Average collective setting with cyclic hubs: 12°

BVWT 067
LONGITUDINAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUNS	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_s}	OBJECTIVE
47	15°	60°	---	0	5000	.30	Cyclic Pitch
43	↓		---	+4°			Control/Tail
42			---	-4°		↓	Off
48	30		---	0		.55	
58	↓		---	+4°			
57			---	-4°		↓	
49	45°		---	0		.81	
59			---	+4°			
71			---	+6°			
68	↓	↓	---	-4°	↓	↓	↓
40	15°	60°	0°	-4°	5000	.30	Stabilizer Con-trol with Cyclic
39			+5°				
41			+10°	↓			
42			---				
45			0°	+4°	↓		
111			+5°		4500		
46			+10°		5000		
43	↓		---	↓		↓	
64	30°		+5°	-4°		.55	
63			+10°				
37/70			+15°				
67			---	↓			
57			+5°	+4°			
56			+10°				
35/61			+15°				
58	↓	↓	---	↓	↓	↓	↓

NOTES:

1. Fuselage angle sweeps
2. Average collective setting with cyclic hubs: 12°

59

BVWT 067
LATERAL/DIRECTIONAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_s}	OBJECTIVE	
94 95	0 ↓	0 ↓	---	0 ↓	5000 ↓	0 .20	Stability. Empennage Off	
96 97	0 ↓	40° ↓					0 .20		
98 99 100	15° ↓	60° ↓					.25 .50 .70		
101 102 103	30° ↓	60° ↓					.55 .70 .81		
104 105	45° ↓	60° ↓	↓	↓	↓	.81 .92		↓

NOTES:

1. Yaw angle sweeps
2. Zero fuselage angle
3. Vertical/horizontal tails off
4. Average collective setting with cyclic hubs: 12°

BVWT 067
LATERAL/DIRECTIONAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_S}	OBJECTIVE
92	0	0	0	0	5000	0	Stability/
93	↓	↓	↓			.20	Empennage On
85	0	40°	0			0	
86	↓	↓	↓			.20	
76	15°	60°	+10°			.25	
77						.50	
78	↓	↓	↓			.70	
87	30°	40°	+15°			.55	
88						.70	
89	↓	↓	↓			.81	
79	30°	60°	+15°			.55	
80	↓	↓	↓			.70	
81						.81	
82	45°	60°	+25°			.81	
83	↓	↓	↓			.92	
84	55°	60°	+35°			.93	↓
90	0	40°	---	↓	↓	0	Stability/"T"
91	↓	↓	---			.20	Horizontal Tail Off

NOTES:

1. Yaw angle sweeps
2. Zero fuselage angle
3. Vertical tail on
4. Average collective setting with cyclic hubs: 12°

BVWT 067
LATERAL/DIRECTIONAL RUN SUMMARY
FORWARD FLIGHT (TRANSITION)

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_s}	OBJECTIVE
106	15°	60°	+10°	+4°	4500	.25	Stability/ with Cyclic Pitch
107	↓	↓	↓	↓		.50	
117				-4°		.25	
118	↓	↓	↓	↓		.50	
108	30°	60°	+15°	+4°		.55	
109				↓		.70	
119				-4°		.55	
120	↓	↓	↓	↓		.70	
110	45°	60°	+25°	+4°		.81	
116			↓	-4°			
172			+35°	+6°			
167	↓	↓	+25°	-6°	↓	↓	↓

NOTES:

1. Yaw angle sweeps
2. Zero fuselage angle
3. Empennage on
4. Average collective setting with cyclic hubs: 12°

BVWT 067
RUN SUMMARY/CROSS COUPLING
OF ROLL CONTROLS AND CYCLIC PITCH CONTROL
FORWARD FLIGHT (TRANSITION)

RUN	i_w	FLAP ANGLE LEFT/RT WING/WING	SPOILER ANGLE LEFT/RT WING/WING	CYCLIC ANGLE	RPM	NOM C_{T_S}
125	15	40°/60°	40°/0	0	4500	.25
123 126 121	30 ↓	40°/60° ↓	40°/0 ↓	+4° 0 -4°	↓	.55 ↓
124 127 122	45 ↓	40°/60° ↓	40°/0 ↓	+4° 0 -4°	↓	.81 ↓

NOTES:

1. Fuselage angle sweeps
2. Stabilizer angle: +20°
3. Average collective setting with cyclic hubs: 12°

BVWT 067
LONGITUDINAL RUN SUMMARY
FORWARD FLIGHT IN GROUND EFFECT
h/D=1.25

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_S}	OBJECTIVE
140	VARY	40°	---	0	4500	.81	Effect of Ground Proximity on Longitudinal Characteristics
141		↓	↓		↓	.70	
142						.55	
143						.25	↓
128		60°	---			.81	
129		↓	↓	↓		.70	
130						.55	
131						.25	↓
149		40°	---	+4°		.81	Cyclic Pitch Control/I.G.E.
150		↓	↓	↓		.70	
151						.55	
152						.25	↓
153		40°	---	-4°		.81	
154		↓	↓	↓		.70	
155						.55	
156		↓				.25	↓
147		40°	+10°	0		.55	Horizontal Tail Effectiveness/ I.G.E.
144		↓	+ 5°	↓		↓	
148	↓	↓	0		↓		
146			-10°				

NOTES:

1. Wing tilt angle sweeps
2. Zero fuselage angle
3. Moving ground plane
4. Average collective setting with cyclic hubs: 12°

BVWT 067
LATERAL/DIRECTIONAL RUN SUMMARY
FORWARD FLIGHT IN GROUND EFFECT
h/D=1.25

RUN	i_w	FLAP ANGLE	STAB ANGLE	CYCLIC ANGLE	RPM	NOM C_{T_S}	OBJECTIVE
132	15°	60°	10°	0	4500	.70	Basic Lat./Dir.
133	↓	↓	↓	↓		.50	Stability I.G.E.
134						.25	
135	30°	60°	15°	0		.81	
136	↓	↓	↓	↓		.70	
137						.55	
138	45°	60°	25°	0		.81	
139	↓	↓	↓	↓		.92	↓
158	15°	60°	10°	−4°		.50	Effect of Cyclic
163	↓	↓	↓	+4°		↓	on Lat/Dir. Stability I.G.E.
159	30°	60°	15°	−4°		.81	
160				−4°		.70	
161				+4°		.81	
162	↓	↓	↓	+4°		.70	
164	45°	60°	25°	+4°		.92	
165				+6°			
166				−6°			
157	↓	↓	↓	−4°	↓	↓	↓

NOTES:

1. Yaw angle sweeps
2. Zero fuselage angle
3. Empennage on
4. Moving ground plane
5. Average collective setting with cyclic hubs: 12°

6.0 TEST RESULTS AND DISCUSSION

The primary objectives of the second test (Phase II) of the full span four prop tilt wing Model VR068Q, which was equipped with cyclic pitch propellers for low speed longitudinal control, were outlined in the Introduction and consisted of the following major topics.

a.   Yaw Control in Hover

b.   Low Speed Descent Performance with Cyclic Pitch

c.   Longitudinal Stability and Control in Transition
        with Cyclic Pitch

d.   Lateral/Directional Stability in Transition
        Including Effect of Cyclic Pitch

e.   Cross-Coupling of Cyclic Pitch with Aircraft
        Surface Controls

f.   Longitudinal Characteristics in Ground Effect
        Including Cyclic Pitch Effectiveness

g.   Lateral/Directional Stability in Ground Effect
        Including Effect of Cyclic Pitch

Data from the subject test can be used in conjunction with the data acquired during the Phase I test of the same model (Reference 1). Also, data obtained during the isolated prop test of a similar prop/hub/nacelle assembly (Reference 2) can be used to determine the effect of the wing/flap system on cyclic pitch control effectiveness.

6.1 YAW CONTROL IN HOVER

Yaw control in hover is achieved by differential wing surface deflections, that is, the double slotted flaps are deflected down on one wing, and the spoilers plus the flaps (if required) are deflected up on the opposite wing. The flap arrangement for downward flap deflections on the right wing is depicted in Figures 6 and 7. Figure 9 shows the spoiler location and up-flap configuration used on the left wing.

The hover mode yaw control testing accomplished during Phase I testing (BVWT 061) and Phase II testing (BVWT 067) was conducted with the leading edge slats retracted, a wing tilt angle of 90°, horizontal tail removed, and the $p^{+,2}$ prop rotation (both props turning down between the nacelles). Data was acquired at five preselected ground heights which varied from an h/D of 4.0 to an h/D of 1.0. The parameter h/D, has been defined as the ratio of the distance between the outboard propeller plane and ground plane to the propeller diameter. Results of the hover yaw control testing are presented in terms of the non-dimensional yawing moment parameter Y.M./ℓT, wherein

Y.M.,	aircraft yawing moment	ft.lbs.
ℓ,	average distance from inboard and outboard propeller $\mathfrak{C}$'s to fuselage $\mathfrak{C}$ (2.653 ft. on the model)	ft.
T,	propeller thrust (total)	lbs.

The hover yaw control power available with the following configurations was measured during the Phase I test.

a. Double slotted flaps deflected down on the right wing, clean left wing.

b. Up-flap deflection on the left wing, clean right wing.

c. Differential flap deflection, flaps down on the right wing and up on the left wing.

d. Spoilers deflected up on the left wing, clean right wing.

During Phase II, yaw control in hover was evaluated with flaps deflected down on the right wing in combination with spoilers on the left wing and also spoilers plus flaps deflected up on the left wing. In addition, the effect of cyclic pitch control inputs on yaw control power was examined.

6.1.1 Hover Yaw Control Power

Figure 24 presents the hover yaw control available with flaps
deflected down on the right wing plus spoilers deflected up on
the opposite wing, along with a comparison of the control power
that would have been available if the individual contributions
of flaps and spoilers were directly additive. The comparison
curve was developed from the flaps alone and spoilers alone
data shown in Figure 24 which was obtained during the Phase I
test (BVWT 061).

Also shown in Figure 24 is the data from a check run made in
BVWT 067 with the double slotted flaps deflected 60° on the
right wing/clean left wing. This data, which is directly
comparable with the data from Run 42 of BVWT 061, reflects an
incremental decrease in hover yaw control power over that pre-
viously measured with the same configuration. An explanation
for this decrease involves disc loading. Yaw control power
testing was conducted in BVWT 061 with the collective hubs
rotating at 6800 RPM. This operating format resulted in an
average out-of-ground effect hover disc loading of 33.4 lbs/ft^2
when a collective blade angle setting of 14" was used. In
contrast, the cyclic hubs used during BVWT 067 allowed a maxi-
mum propeller operating speed of 5000 RPM. This model config-
uration provided an equivalent disc loading of 16.7 lbs/ft^2
with 12.5° of collective. It is reasonable to assume that
the decrease in Reynolds number associated with the reduction
in disc loading resulted in lower flap turning effectiveness
and consequently a lower yawing moment capability.

Further, it would be anticipated that Reynolds number would not
influence spoiler turning effectiveness to the degree that it
effected flap turning effectiveness. Thus it appears that
some loss in yaw control power occurred when down flaps/right
wing were tested along with up spoiler/left wing over that
that can be attributed to Reynolds number effects,ie, in
Figure 24 compare the average increment of h/D's of 4.0 and 2.0
between the two differential control surface lines and the
increment between the two 60° flaps alone lines, the increment
being considerably larger for the former case. This noted
loss is discussed later.

Figure 25 illustrates the change in yawing moment capability
when up flaps were used in combination with spoilers on the
left wing. The incremental improvement obtained out-of-ground
effect diminished to zero in-ground effect. As a consequence,
an up-moving flap does not appear to have sufficient effective-
ness to warrant its use as a yaw control device in hover when
a spoiler is already being considered.

The Phase I test report (Reference 1) discussed the apparent anomaly wherein the measured yawing moment capability decreased between ground height values (h/D) of 2.0 and 4.0. In actuality an out-of-ground effect condition exists at h/D's greater than 2.0, and it would be expected that the measured yawing moment would not change between values of h/D of 2.0 and 4.0. Wires with long wool tufts attached were strung across the test section in front of the model to investigate the possiblity of flow recirculation in the test area. No noticeable tuft movement could be discerned during the vertical traverses of the model. Even though this check was negative, some amount of flow recirculation could be present, however, it would be necessary to employ the use of an accurate anemometer and take measurements at several locations around the model and at a couple of model heights to completely investigate the possibility.

Model dynamics was noted in the Phase I test report as a possible source of anomaly. In the hover testing, it was observed via oscilloscope presentations that model dynamics, which manifested itself in the oscillatory nature of the data, was most severe at a model height of 2.0 h/D.

As pointed out on Page 80 of the Phase I test report (Reference 1), the necessary value of the non-dimensional parameter Y.M./ ₤T is 0.293 for a representative tilt wing transport-type aircraft hovering at its design "V" gross weight of 86,930 lbs and meeting a 0.5 radian/sec^2 yaw angular acceleration level.

6.1.2 Hover Download due to Yaw Control

Figure 26 presents the loss in vertical lifting capability associated with the use of wing control surfaces for hover yaw control. The download shown in this figure was calculated as a percentage of the total propeller thrust as measured at each data point.

$$\text{Hover Download, } \%T = \left(1 - \frac{\text{Normal Force}}{T}\right) 100$$

Figure 26 shows the download as a function of ground height for the following configurations tested during BVWT 067: down flaps on one wing alone, down flaps (right wing) plus up spoilers (left wing), and down flaps (right wing) plus up spoilers and flaps (left wing). Comparative data presented in Figure 26 with dashed lines faired through the points was obtained in BVWT 061. The hover download data from BVWT 061 is comprised of the clean wing, up spoilers on one wing alone and down flaps on one wing alone configurations. In each case tested, the download varies from a maximum for the out-of-ground effect condition to a net upload at the lowest model

69

height evaluated (h/D=1.0). This upload is probably a result of the positive pressures being generated on the bottom of the fuselage.

One item of interest that is apparent from an examination of Figure 26 is that the incremental download due to the spoilers deflected up on one wing plus the incremental download due to the flap deflected down on the opposite wing is less than that measured for the combined yaw control configuration. It was previously observed that the yaw control capability with the combined control was less than that measured for the individual parts. An explanation for this situation is that transverse flow from one side of the aircraft to the opposite side, occurred with the differential control configuration. The transverse flow reduced the yawing moment capability and at the same time increased the download as a result of higher negative pressures on the bottom of the fuselage.

The overall measured download of 14.2% for the 60° spoiler/left wing plus 60° flap/right wing configuration (Run 12) is high with respect to the 11.7% download previously measured for the 40° up flap/left wing plus 60° flap/right wing configuration and presented in Figure 45 of the Reference 1 test report. Hover download measurement is extremely sensitive to the accuracy of the data. A check was made of the balance calibrations used for the reduction of the BVWT 067 Phase II data and these prime sensitivities were compared to those used for reducing the BVWT 061 Phase I data. This exercise showed the propeller thrust data from the Phase II test to be 2.5% higher in magnitude than the corresponding data recorded from the Phase I test as a result of calibration differences. Calculations established that the downloads registered during Phase II were 2.2% larger than comparative download data from Phase I. A download adjustment of 2.2% to the Run 12 data would bring the download for this run (12% instead of 14.2%) in line with Phase I data, however, a download adjustment of this magnitude is probably on the high side; i.e., compare in Figure 26 the 60° flap deflection/right wing alone run from BVWT 067 to similar data from BVWT 061. The download from Phase II (5000 RPM) should have been somewhat lower than the download from Phase I (6800 RPM) to be consistent with the corresponding yawing moment data comparison presented in Figure 24, but not excessively less.

The above investigation also indicates that the non-dimensional yawing moment (Y.M./ℓT) measured during Phase II was 2.5% relatively lower in magnitude than the values recorded from the Phase I test.

During the analysis of the download data from BVWT 067, it was noticed that the propeller thrusts at an h/D of 4.0 were

1.2% to 2% higher than those measured at an h/D of 2.0 during Runs 12 and 13. This difference which does not appear to be reasonable, increased the download accordingly at an h/D of 4.0.

6.1.3 Effect of Cyclic Pitch on Hover Yaw Control

Two runs were performed during BVWT 067 to evaluate the effect of cyclic pitch inputs on hover yaw control capability. In this investigation, a yaw control configuration of spoilers deflected 60° on the left wing plus flaps deflected 60° on the right wing was selected for the evaluation. Cyclic pitch inputs of +6° were used. The results from the testing are presented in Figure 27.

It can be seen that positive cyclic (nose down pitching moment) increases the yaw control power while negative cyclic (nose up moment) produces an opposite effect. This variation is a result of the location of the propeller centerlines (both inboard and outboard) below the wing chord plane in combination with the shift in the center of pressure cutting across the propeller disc due to cyclic pitch action. Positive cyclic shifts the center of pressure closer to the wing chord plane thus increasing the dynamic pressure acting over the wing. Negative cyclic has an opposite effect - a shift in the center of pressure further away from the wing.

The hover download data presented in Figure 28, a companion plot to Figure 27, corroborates this explanation. The increase in hover download with positive cyclic pitch is consistent with the improvement in yaw control capability with positive cyclic.
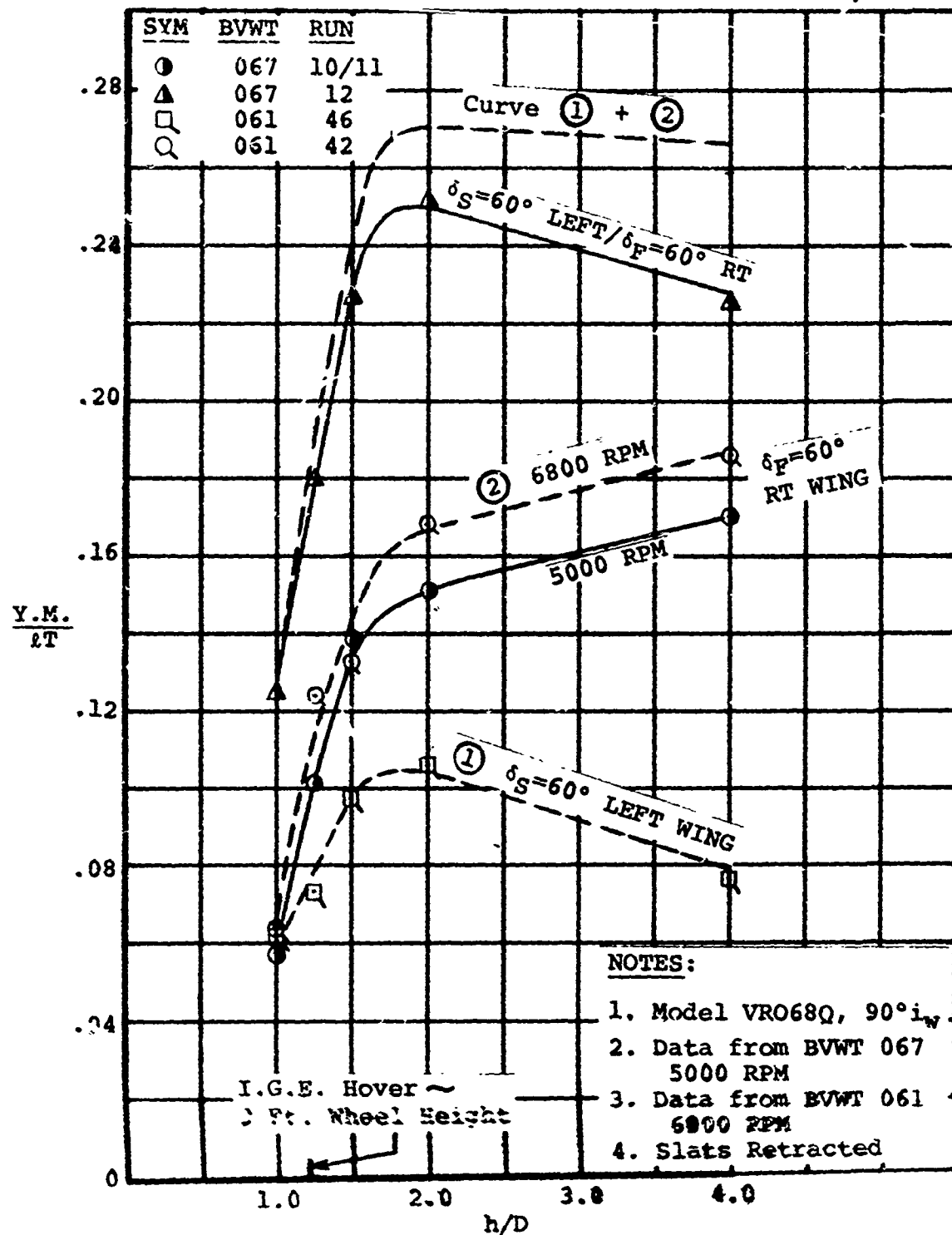
HOVER YAW CONTROL BUILDUP WITH
FLAPS (RT WING) AND SPOILERS (LEFT WING)

72

Figure 25

$\frac{Y.M.}{\ell T}$

$\delta_F = 60°$ RT
$\delta_S = 60°$ & $\delta_F = -40°$ LEFT

$\delta_S = 60°$ LEFT/$\delta_F = 60°$ RT

SYM	RUN
△	12
◊	13

NOTES:
1. Model VRO68Q, 90° i_w
2. Data from BVWT 067
3. 5000 RPM
4. Slats Retracted

I.G.E. Hover ~
2 Ft. Wheel Height

h/D
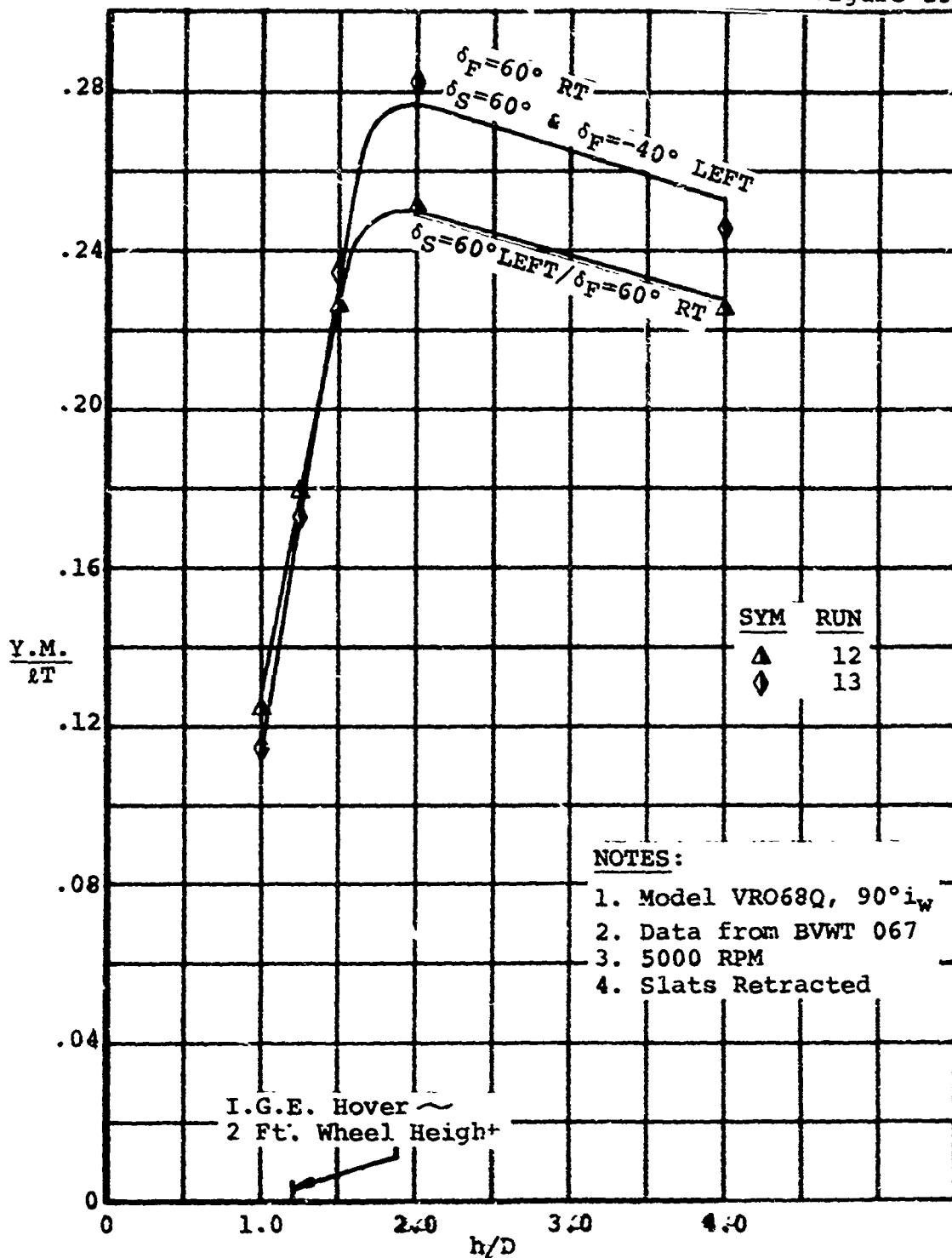
HOVER YAW CONTROL
DOWN FLAPS ~ RIGHT WING
UP FLAPS AND/OR SPOILERS ~ LEFT WING

73

Figure 26

SYM	BVWT	RUN
◑	067	10/11
△	067	12
◈	067	13
✕	061	34
◻	061	46
◯	061	42

$\delta_F = 60°$ RT
$\delta_S = 60°$ & $\delta_F = -40°$ LEFT

Estimate
6800 RPM

$\delta_S = 60°$ LEFT/$\delta_F = 60°$ RT

6800 RPM

$\delta_F = 60°$ RT WING

5000 RPM

$\delta_S = 60°$ LEFT WING

CLEAN WING

%T

h/D

NOTES:
1. Model VRO68Q, 90° i_w
2. Data from BVWT 067
   5000 RPM
3. Data from BVWT 061
   6800 RPM
4. Slats Retracted
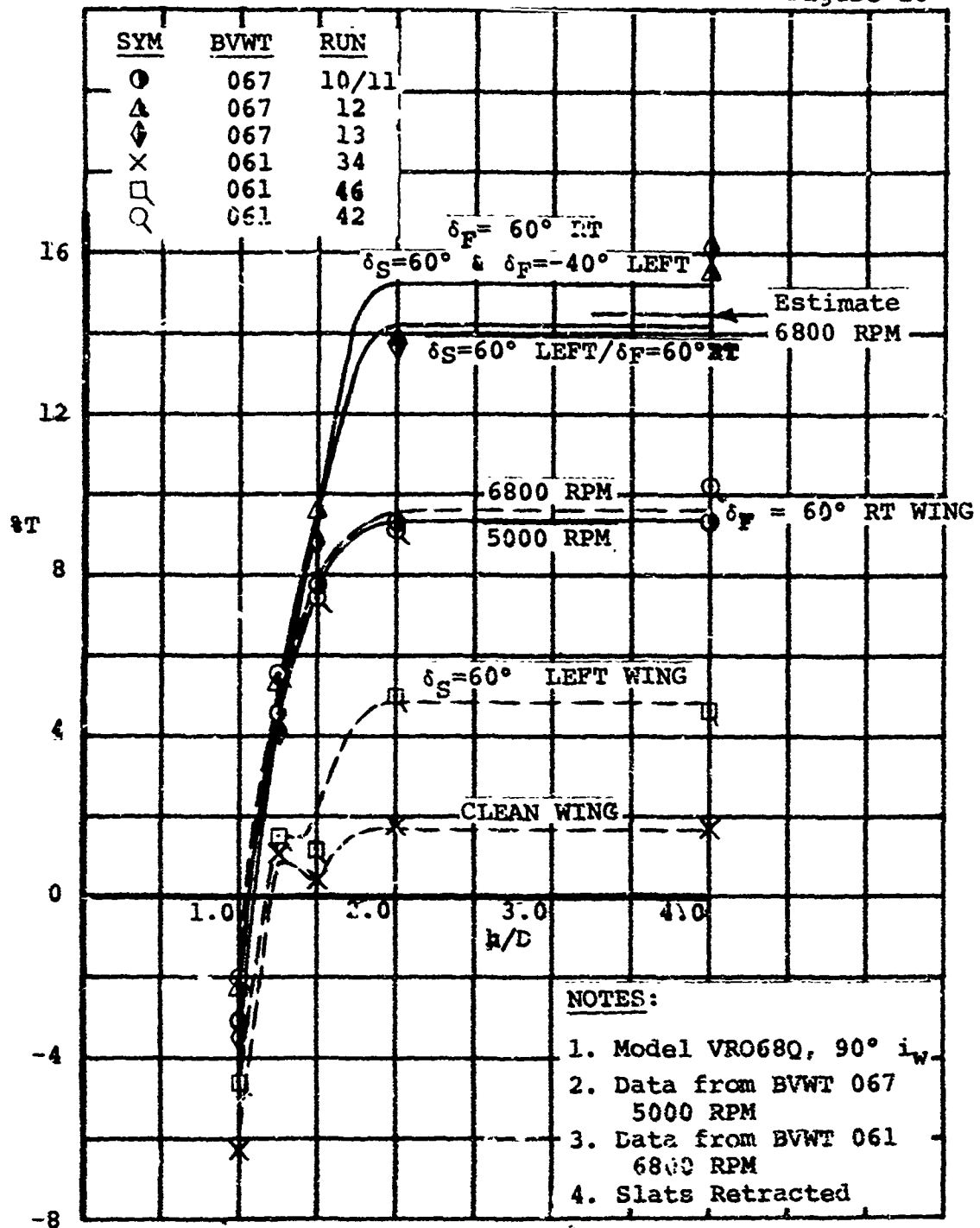
HOVER DOWNLOAD DUE TO
FLAP AND SPOILER DEFLECTIONS

74
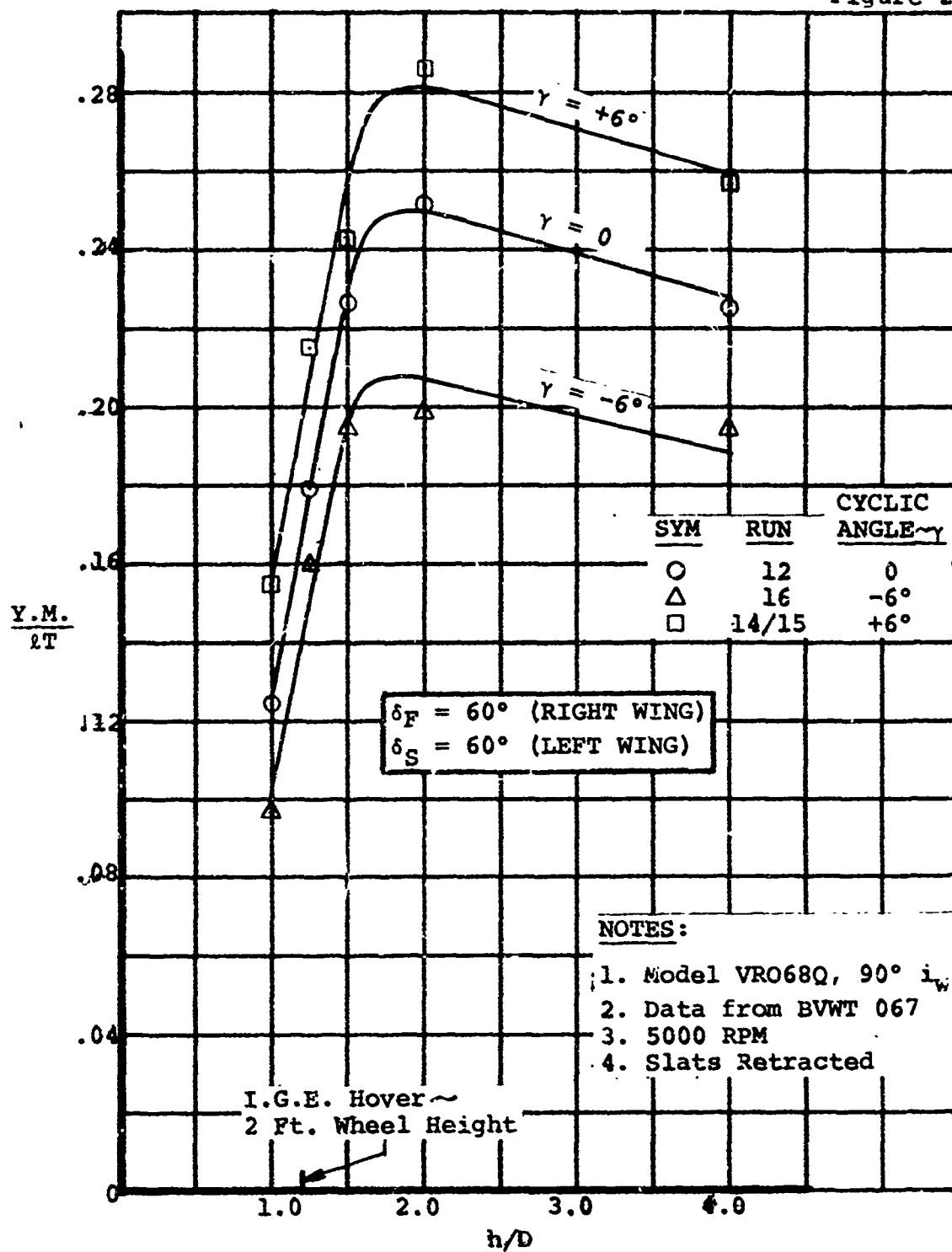
EFFECT OF CYCLIC PITCH
ON HOVER YAW CONTROL

Figure 28
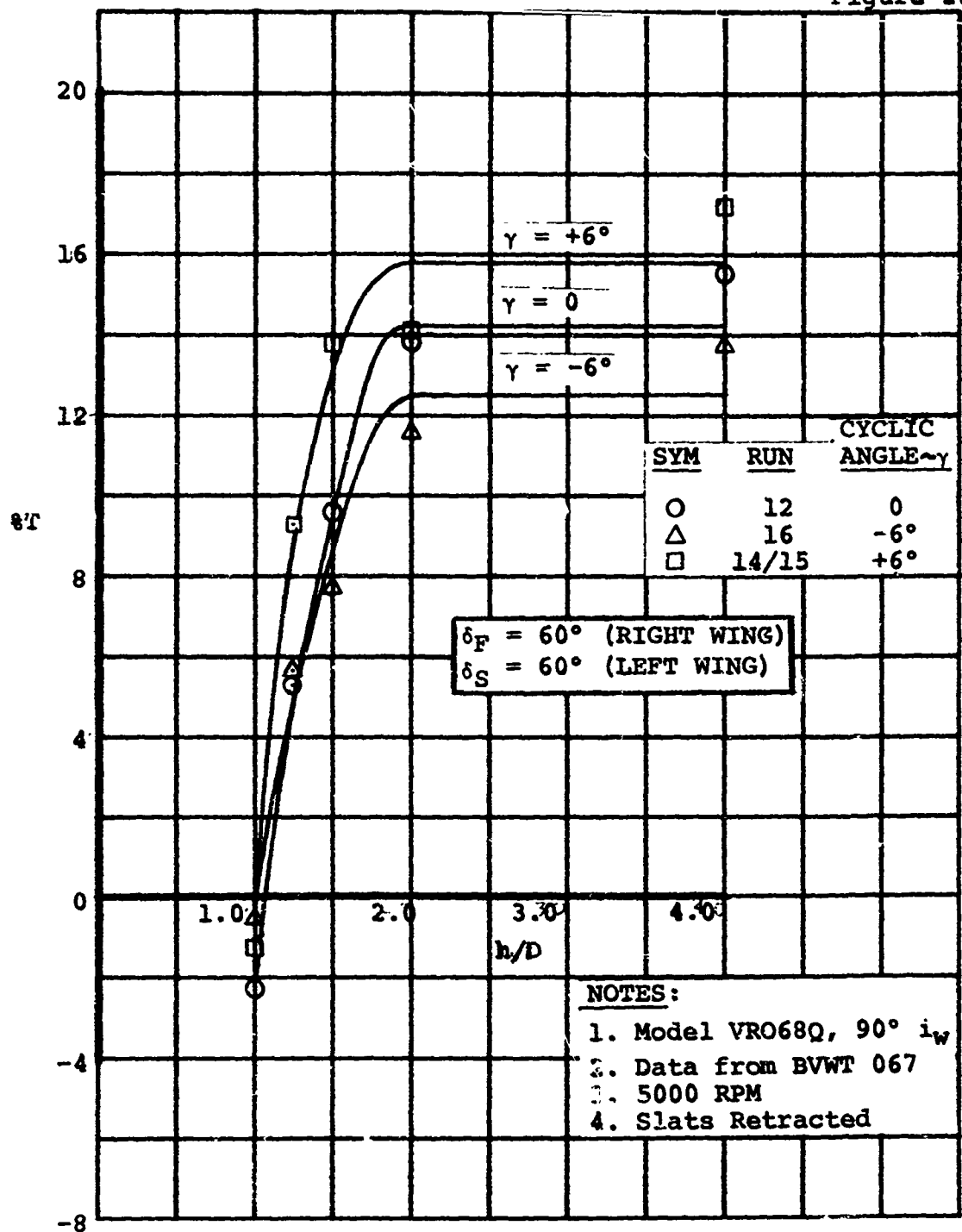
EFFECT OF CYCLIC PITCH ON
HOVER DOWNLOAD DUE TO YAW CONTROL

6.2 CYCLIC PITCH CONTROL IN HOVER

6.2.1 Effect of Horizontal Tail on Cyclic Pitch Control

The effectiveness of the propellers in producing aircraft pitch-
ing moment when cyclic pitch is applied was investigated in the
hover mode during Phase I test BVWT 061. Since this testing
was conducted with the horizontal tail removed, a run was made
during BVWT 067, (varying ground height in hover) with the
horizontal tail on in order to investigate possible pitching
moment interference effects between the propellers and the
tail.

A cyclic pitch angle of +6° was selected for the run as this
case would maximize the downwash in the regime of the horizon-
tal tail thus maximizing potential interference effects on
aircraft pitching moment. The horizontal tail was set at a
typical stabilizer angle for hover (+45°).

Figure 29 presents the results of this investigation. Depicted
on the chart is the fairing for cyclic pitch control effective-
ness derived from BVWT 061 (Reference 1) with the horizontal
tail off. This line is applicable to an out-of-ground effect.
hover condition (h/D's of 4.0 and 2.0). It can be noted that
the horizontal tail on data measured during BVWT 067 with a
cyclic angle of +6° agrees well with the previously derived
"tail off" fairing and the data exhibits only a small variation
with ground height, leading to the conclusion that the hori-
zontal tail has not appreciable effect on aircraft pitching
moment in the hover mode.

$$\frac{M}{D \times T} = \frac{C_{M_P}}{C_T}$$

NOTES:
1. Model VRO68Q, 90° i_w clean config.
2. Data from BVWT 067 (Run 17)
3. 5000 RPM
4. Moments about wing pivot

Horizontal Tail Off Fairing/BVWT 061 (See Figure 28/ Reference 1)

h/D=4.0 & 2.0

Cyclic Angle, $\gamma \sim$ Deg.

SYM	h/D
○	4.0
□	2.0
△	1.5
◇	1.25
○	1.0

Horizontal Tail On
$\measuredangle$ =+45°
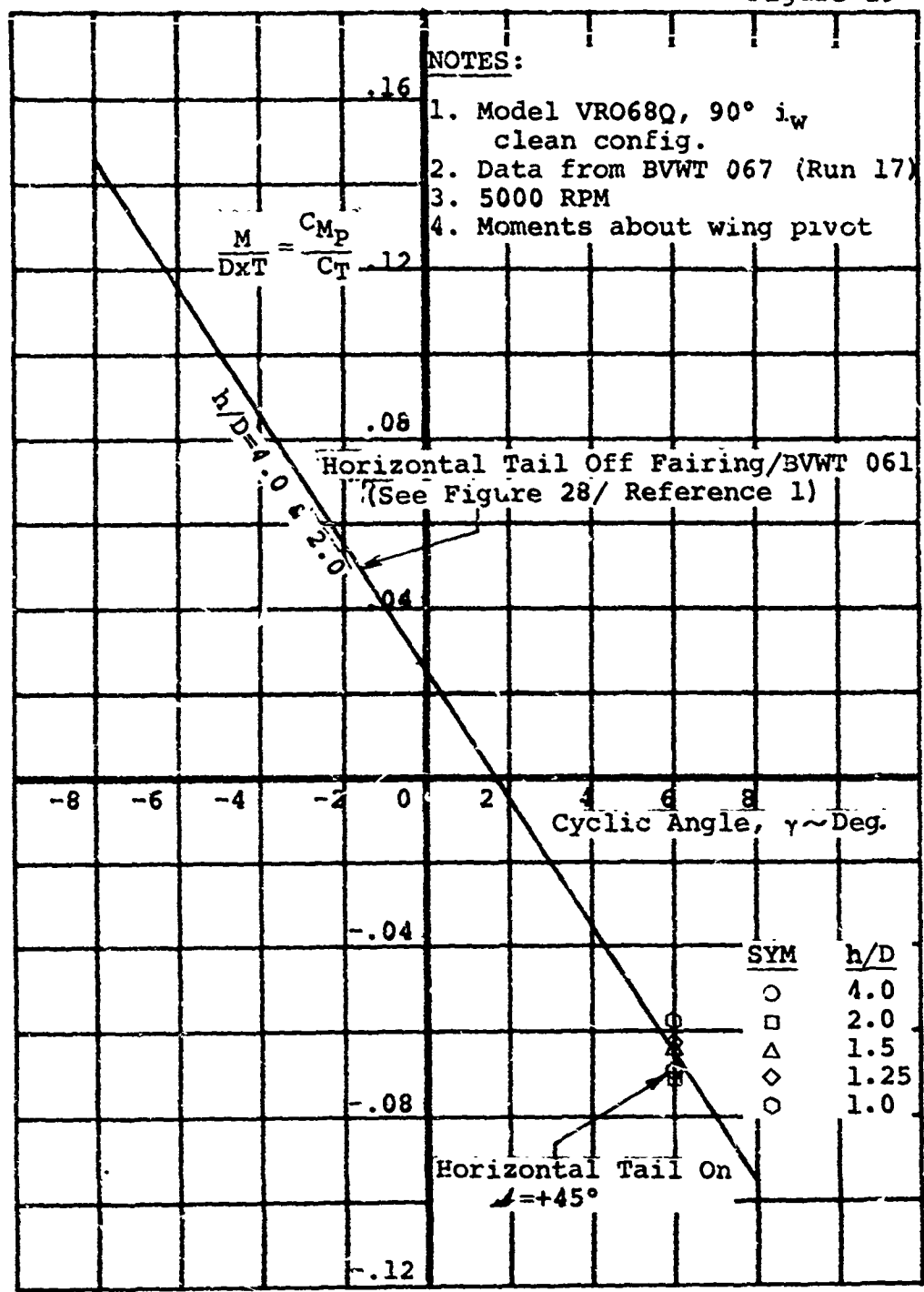
EFFECT OF HORIZONTAL TAIL ON PITCHING MOMENT DUE TO CYCLIC VARYING GROUND HEIGHT IN HOVER

6.3 LOW SPEED DESCENT PERFORMANCE

One of the critical design items for a tilt wing aircraft is
the low speed descent capability. (Boeing-Vertol has been
using as a design goal, a minimum descent rate of 800 fpm up
to a flight speed of 42.5 kt and a descent angle of 12° at
higher speeds.)

The leading edge slat and double slotted flap configurations
utilized on Model VRO68Q, were developed during the 1969
Boeing-Vertol wind tunnel test program of a four prop tilt
wing aircraft as a means of meeting the descent rate goal. In
the same test program, the placement of the propeller hub $\underline{\mathfrak{c}}$'s
with respect to the wing leading edge, was investigated with
regard to maximizing descent performance. Test results were
incorporated into the design of Model VRO68Q. The $P^{1,2}$ prop
rotation direction (props turning down between the nacelles)
used during the two phase test program on Model VRO68Q was
also selected on the basis of previous wind tunnel tests to
maximize descent capability.

The low speed descent capability testing concerned with cyclic
pitch effects was commenced during the Phase I test (BVWT 061)
with a wing configuration consisting of double slotted flaps
deflected 60° and full span slats positioned to the $Q^{*10,10,*}$
setting. In this setting, each slat segment located behind
an up-turning propeller is deflected to a higher angle. During
these test runs investigating the reduction in descent capabi-
lity with positive cyclic angles, damage occurred to the
cyclic hubs, necessitating a delay in the completion of this
investigation to the Phase II test.

6.3.1 Effect of Cyclic Pitch on Descent Capability

The results of the testing during BVWT 061 and BVWT 067, that
investigated the effect of cyclic pitch on descent performance
are presented in Figure 30. Descent rate at buffet onset is
shown as a function of full scale aircraft velocity, V_F. As
discussed in the Data Reduction section, V_F and R/D were calcu-
lated using a wing loading of 73.5 psf and test section at-
mospheric conditions. It was necessary for data consistency
between the two tests, to adjust the Phase II data for the
difference between the test section air density occurring in
the Phase II test and that prevailing during the Phase I test
($\rho=.00228$ slugs/ft^3).

Descent capability at buffet onset was determined by establish-
ing the wing incidence angle at which initial stall or separa-
tion occurred on the wing outboard of the inboard set of
fences, i.e. outboard of the region between fences shown in

79

Figure 2. Inboard areas where stall was tolerated comprise
the wing center section over the fuselage, the area
behind the gap between the propeller tip and fuselage side,
and the region between the fences - sections over which the
low freestream q or less than full slipstream q prevail and
roll disturbances are minimal.

In selecting the buffet onset angle, tuft photographs and
observer written comments were studied in conjunction with
corresponding force polars presented in terms of L/qb^2 vs
X/qb^2 rather than the slipstream notation C_{L_S} vs C_{X_S}. With
polars in the L/qb^2 vs X/qb^2 format, initial stall coincides
with a definite break in the curve at thrust coefficients up
to 0.65 C_{T_S} or speeds greater than 55 knots. At lower speeds,
the stall buildup is gradual and at a wing angle 10° beyond
that selected for buffet onset, the flow remains attached on
75% of the wing.

Another curve that has been found useful in determining the
descent performance is the variation with C_{T_S} of the effective
wing angle of attack, $\alpha_{W_{EFF}}$, at buffet onset. (The latter para-
meter is described in the Data Reduction section.) This curve
and the companion plot of wing incidence angle at buffet onset
vs C_{T_S} should be reasonably smooth and consistent as can be
noted in Figure 31.

Previous Boeing-Vertol tilt wing testing has shown that posi-
tive cyclic angles (nose down pitching moment) reduces descent
capability and negative cyclic improves it. In line with
these results only positive cyclic angles were evaluated in
the subject test program. Figure 30 shows that the loss in
descent rate due to positive cyclic averages 100 fpm per de-
gree of cyclic at 62 knots with a reduction in the average to
50 fpm per degree of cyclic at a flight speed of 38 knots.
This reduction in the loss rate at low speeds is consistent
with the hover yaw control testing (discussed previously) that
shows positive cyclic to be beneficial and negative cyclic to
have the reverse role.

A large portion of the loss in descent performance with posi-
tive cyclic has been associated with a reduction in the slip-
stream turning effectiveness. Evidence of this is inclusive
in Figure 31 which presents the buffet onset angles correspond-
ing to the descent data plotted in Figure 30. Buffet onset
angles are shown as wing angles of attack at initial stall and
as the calculated effective wing angles of attack defined as
the angle between the wing chord and the resultant velocity
vector at the wing. At 0.77 C_{T_S} (49 knots) the 4° decrease in
wing buffet onset angle resulting from the application of 6° of

80

positive cyclic represents only 175 fpm out of the total 500 fpm
loss in descent performance. The remainder of the loss, 325
fpm, reflects the reduction in turning effectiveness.

Note that Figure 31, which presents data from wing sweep runs
with the fuselage set at 0°, expresses buffet onset angles in
terms of wing incidence angle plus fuselage angle ($i_w + \alpha_F$).
This was necessary in order to account for the incremental wing
angle of attack being produced by the upward deflection of
the model on the sting due to the high lift forces generated.
The increment in angle of attack from sting deflection is
accounted for by a calculation routine in the data reduction
program.

Figures 32 through 37 present the basic data from the runs
that were conducted for the purpose of determining the low
speed descent capability. The data is presented as plots of
L/qb^2 vs X/qb^2 (force polars), L/qb^2 vs i_w (lift curves), and
and M/qb^2c vs i_w (moment curves). Marked off on each force
polar plot are the selected buffet onset points and a .3g de-
celeration line (corresponds to 16.7° descent angle) for re-
ference purposes. Figure 38 depicts the variation of slip-
stream thrust coefficient prevailing during the +4° cyclic
runs with 5000 RPM and an average collective setting of 12.8°.

81
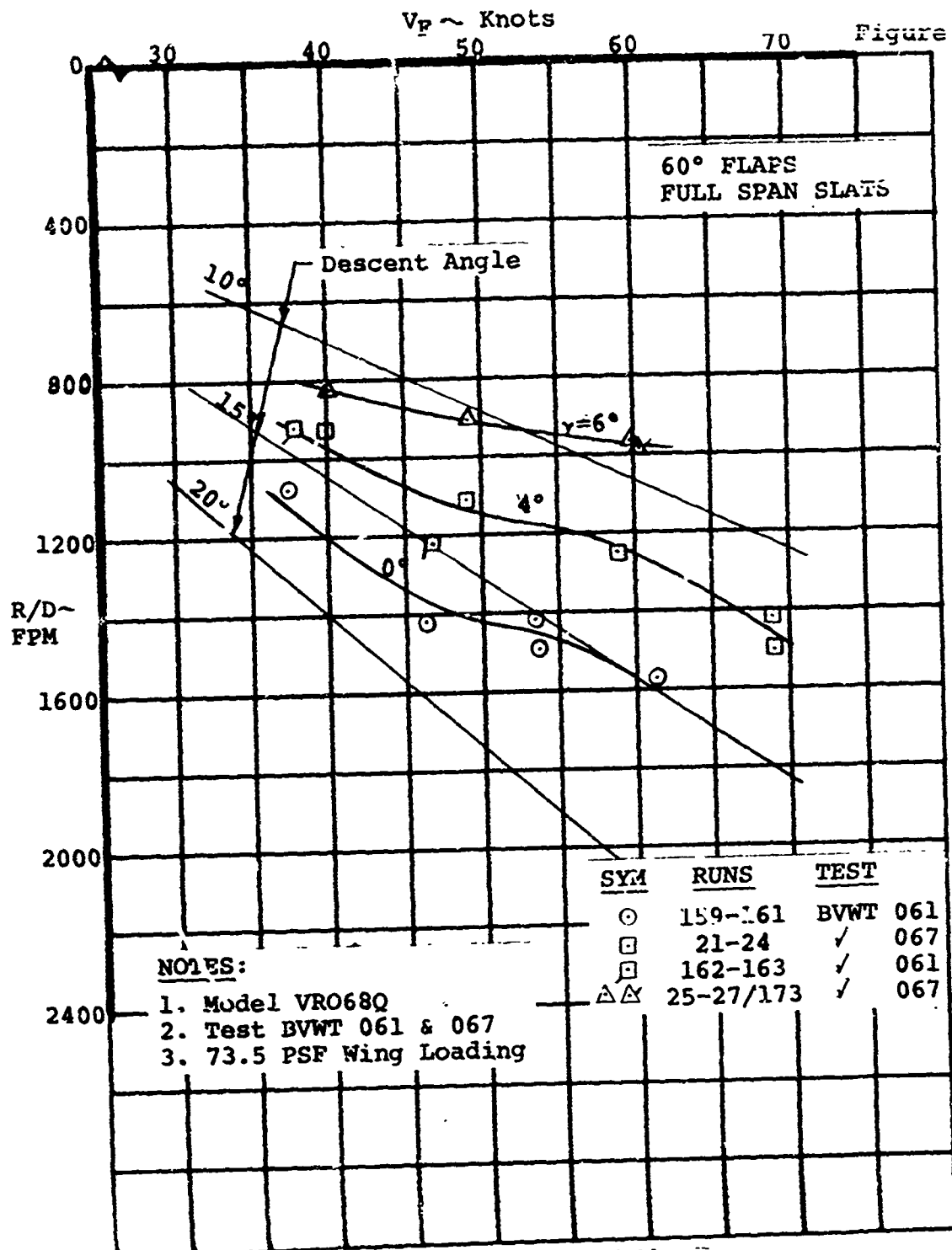
EFFECT OF CYCLIC PITCH ON
LOW SPEED DESCENT CAPABILITY

Figure 31
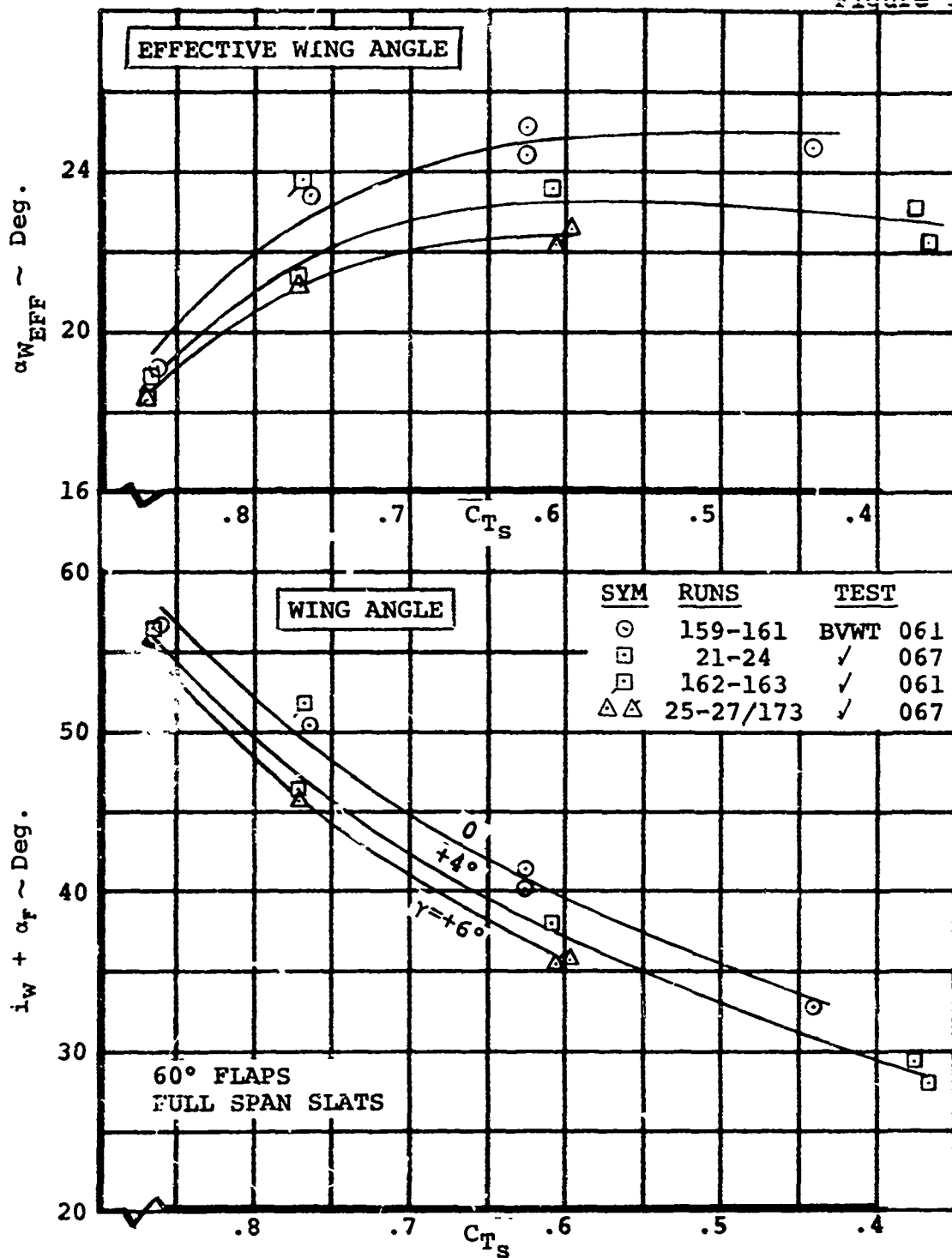
EFFECT OF CYCLIC PITCH ON
BUFFET ONSET ANGLE

83
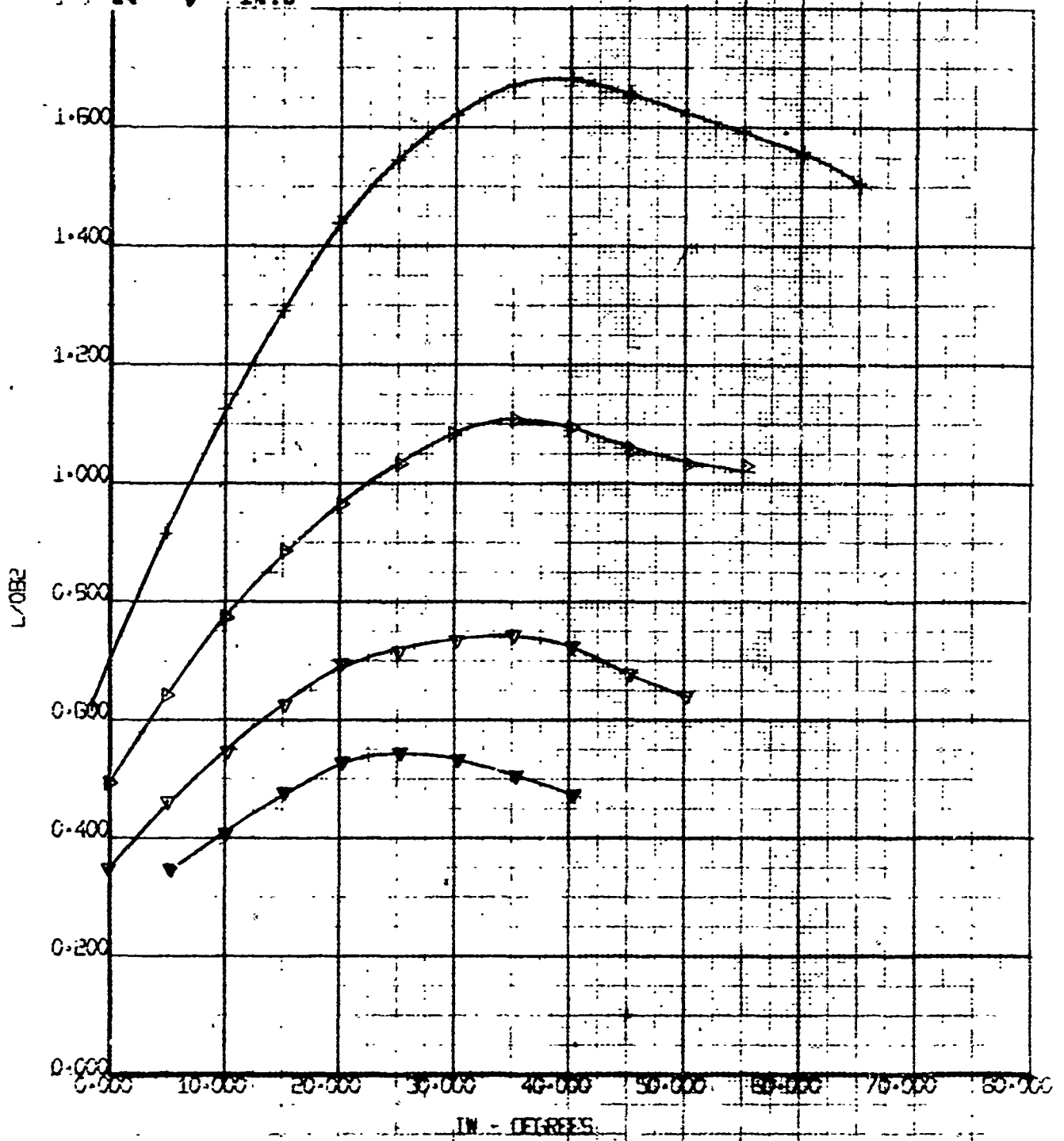
DESCENT CAPABILITY RUNS
$\delta_p = 60°$, FULL SPAN SLATS
+4° CYCLIC

RUN	SYM	q
21	+	2.76
22	▷	4.70
23	▽	8.49
24	▼	14.0

FOUR-PROP TILT WING
MODEL VZ0680 (FULL SPAN)
L/QB2 VS. X/QB2

BVWT
67

1/4/71

SHEET 84

Figure 33. DESCENT CAPABILITY RUNS, $\delta_F = 60°$, FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
L/qB^2 VS. i_W

NUMBER D170-10039-1

BVWT 67

SHEET 85

DESCENT CAPABILITY RUNS

CYCLIC

RUN	SYM	q	RPM
25	+	2.76	5000
26	▷	4.70	
27	▽	8.49	
173	▼	6.20	4500

L/QB2

1·800
1·600
1·400
1·200
1·000
0·800
0·600
0·400
0·200
0·000

0·000   10·000   20·000   30·000   40·000   50·000   60·000   70·000   80·000

IW - DEGREES

FOUR-PROP. TILT WING	BVWT
MODEL VZ0GRO(FULL SPAN)	67
L/QB2 VS. IW	12/7/70

NOT REPRODUCIBLE

SHEET 87

RUN	SYM	q	RPM
25	+	2.76	5000
26	▷	4.70	✓
27	▽	8.49	✓
173	▼	6.20	4500

DESCENT CAPABILITY RUNS
$\phi_F = 60°$, FULL SPAN SLATS
+6° CYCLIC

Buffet Onset

-3g

i_w by 5°

$i_w = 0°$

L/QB²

X/QB²

FOUR-PROP TILT WING		BVWT
MODEL VROBSA (FULL SPAN)		67
L/QB² VS. D/QB²		12/ 7 '70

DESCENT CAPABILITY RUNS
$\delta_F = 60°$, FULL SPAN SLATS
+ 6° CYCLIC

RUN	SYM	q	RPM
25	+	2.76	5000
26	▷	4.70	
27	▽	8.49	
173	▼	5.20	4500

IW - DEGREES

PM/QB²C

FOUR-PROP TILT WING
MODEL V20660 (FULL SPAN)
PM/QB²C VS. IW

BVWT
6'

12/7/70

NOTES:
1. Model VR068Q
2. Data from BVWT 067
3. Full Span Slats
4. Horizontal Tail Off
5. Zero Fuselage Angle
6. Cyclic Angle =+4°

Figure 18

RUN	SYM	q
21	○	2.76
22	□	4.70
23	△	8.49
24	◇	14.0

NOM C_{T_S} = .85
NOM C_{T_S} = .75
NOM C_{T_S} = .58
NOM C_{T_S} = .37

i_w ~ DEGREES

VARIATION OF C_{T_S} WITH WING INCIDENCE
δ_F=60°
5000 RPM

SHEET 90
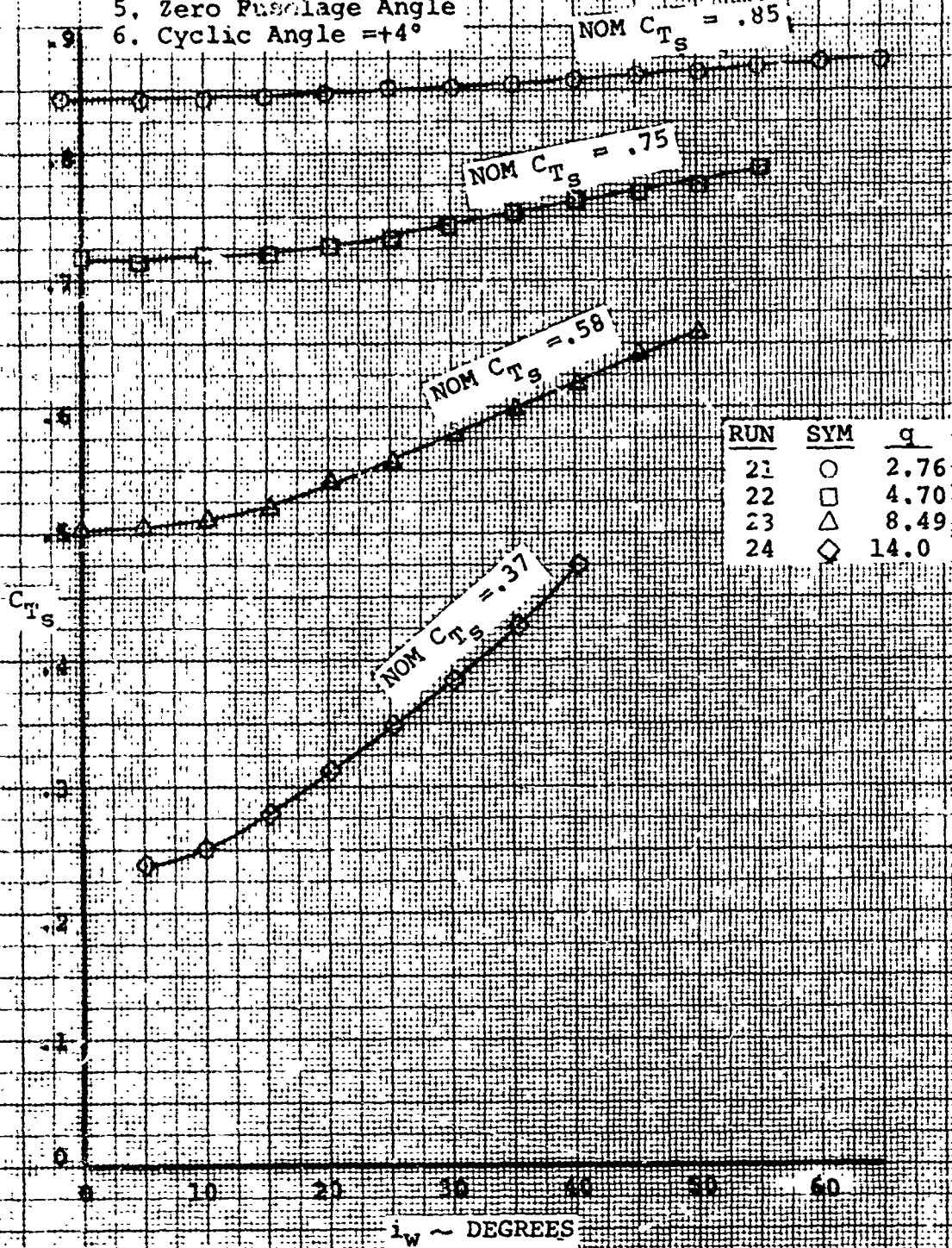
6.4 LONGITUDINAL STABILITY AND CONTROL IN TRANSITION

Longitudinal control with cyclic pitch and its effect on the
basic aircraft longitudinal characteristics were investigated
through the transition flight regime with and without the
horizontal tail depicted in Figure 16. High positioning of
this tail on the vertical fin as illustrated in Figure 15 re-
sulted in a horizontal tail volume coefficient of 1.33. The
potential influence of cyclic pitch inputs on the horizontal
tail effectiveness was also evaluated.

Fuselage pitch runs were performed at constant RPM for selec-
ted combinations of wing tilt angle and flap angle that were
representative of typical combinations required in transition.
Data was taken at wing incidence settings ranging from 15° to
55° with 60° of flap angle. The slipstream thrust coefficient
(C_{T_S}) for these runs varied from 0.3 for the 15° wing tilt
angle case to 0.93 for the 55° wing tilt angle case. Runs
were made with a sufficient number of cyclic pitch angular
settings and stabilizer settings to meet the test objectives.

As mentioned in Section 2.7 of this report, the wing angle of
attack range for evaluation with a prescribed combination of
wing tilt angle and thrust coefficient was adjustable per the
sting "pre-bend" angle selection. These "pre-bend" angles
were chosen prior to each set of runs so that the data could
be acquired at or near representative flight conditions.

All moments obtained during the longitudinal stability runs
were transferred to a representative mid-center of gravity.
This center of gravity moves up and back as the wing tilt
angle increases. Positions of the c.g. for the various wing
tilt angles used, were calculated by utilizing the scaled wing-
down longitudinal and vertical locations for the fixed mass
c.g. and rotating mass c.g. (with respect to the wing chord
line) of a representative transport-type four prop tilt wing
aircraft with a V-mode gross weight of 86,930 lb.(and no di-
hedral), plus the wing pivot location of the model. Model
scale with respect to this aircraft was 1/12.3.

Figure 39 shows the relative locations of the rotating mass
c.g., fixed mass c.g., and resultant aircraft c.g. to the
wing pivot for the wing down case and 70° wing tilt case. The
movement of the aircraft c.g. with wing tilt angle that was
used in the data reduction program is illustrated in Figure 40.
As can be seen, the movement of the c.g. for a wing angle
change from zero to 70° is 10.3% MAC aft and 15.5% MAC up.

$i_w=0°$

Rotating Items C.G. (.558W)

24.0% z
13.7% x

Wing Pivot

Total A/C C.G. (W)

0
25   x ~ % m.a.c.

-30.3% z
39.3% x

Fixed Items C.G. (.442W)

Balance Center

$i_w=70°$

Wing Chord Line @ $i_w=0°$

Wing Pivot

15.5

0
35.3   x ~ % m.a.c.

Balance Center
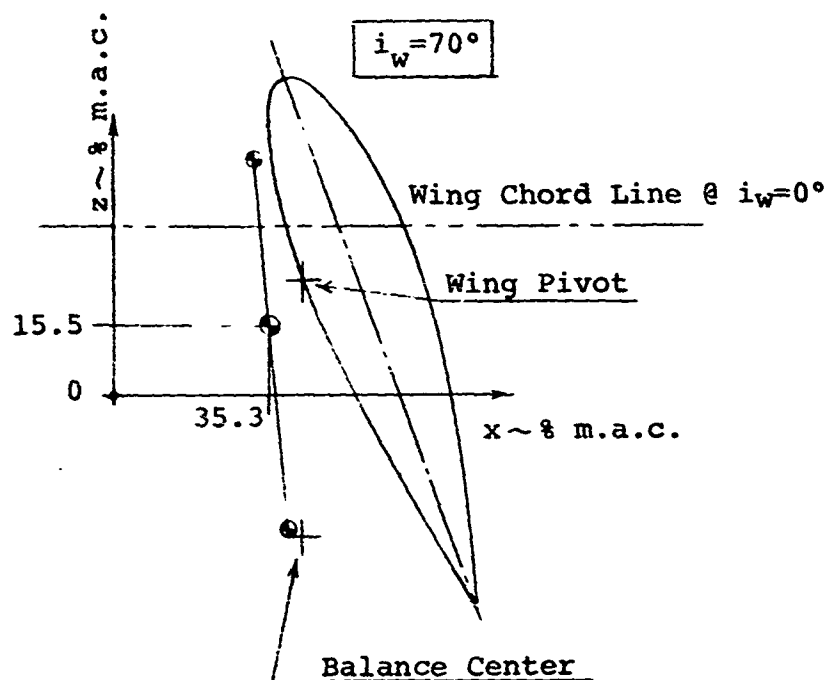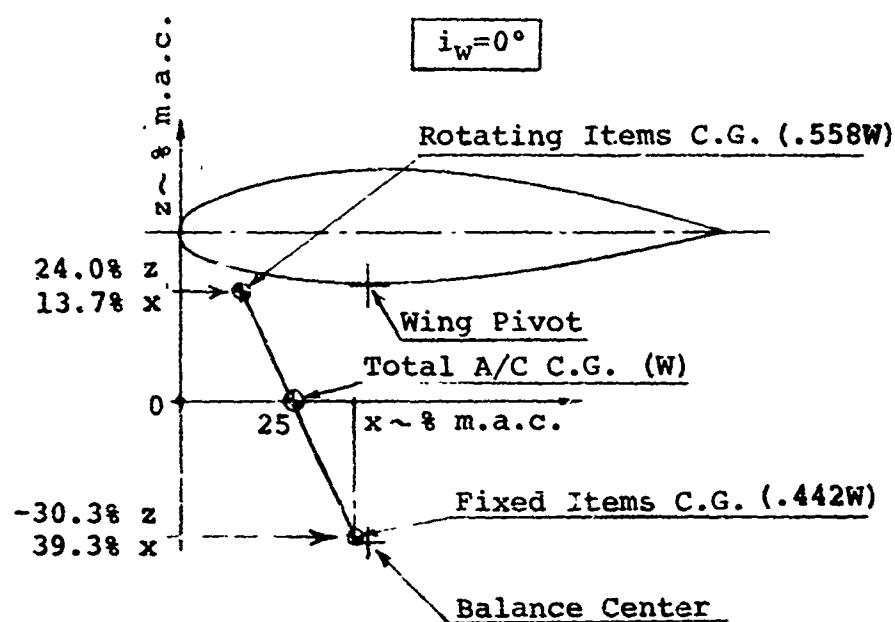
SCHEMATIC RELATIONSHP BETWEEN
FIXED AND ROTATING PORTIONS OF A/C WEIGHT
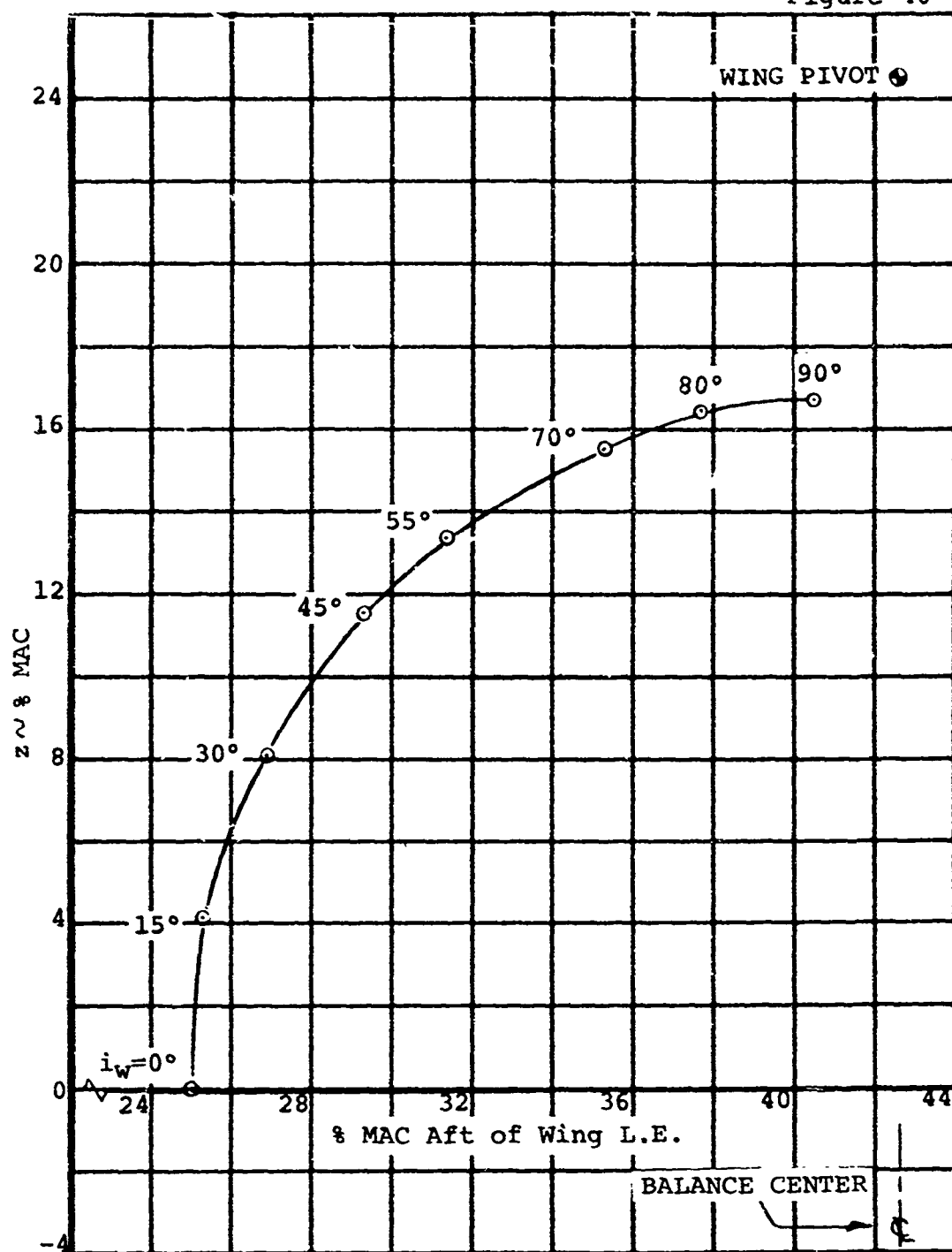
MOVEMENT OF A/C CENTER OF GRAVITY
WITH WING TILT

6.4.1 Cyclic Pitch Control Effectiveness

Of concern in the investigation of the control capability of installed cyclic propellers was the large increase in basic hub pitching moment produced at high propeller shaft angles or wing tilt angles with the high lift system extended, a condition experienced in descent. Figure 147 of Reference 1 shows that the initial rate of build-up of hub pitching moment on the installed propeller is about three times that of the isolated propeller for the various advance ratios (J) tested with the full span slats extended and flaps deflected to 60°. However, the curves, especially for the outboard props appear to be bending over or "peaking" at much lower angles of attack than the isolated prop. The cyclic pitch control testing performed on the full span model was directed towards determining the possible wing/flap effects on cyclic pitching moment characteristics.

Figure 41 presents a comparison in slipstream coefficients of the pitching moment capability per degree of cyclic obtained on the full span model with those measured on the 1/12th scale isolated propeller model at directly comparable conditions of 5000 RPM and 12° of blade angle. The data presented, is for shaft angles corresponding to essentially constant descent angles over the thrust coefficient range. As indicated, higher shaft angles, which are within 5° of the respective wing tilt angles in this presentation, are required at the larger thrust coefficients or lower forward speeds. The descent case was selected for this presentation, since the combination of shaft angle and wing configuration, full span slats deployed and 60° of flap deflection, would potentially have the largest adverse effect on cyclic pitch effectiveness, due to the increased basic hub pitching moment and possible flow distortion.

Except for the lowest thrust coefficient condition tested (.32 C_{T_S}), the cyclic pitch effectiveness measured about a mid c.g. of the full span model was larger than that recorded at the hub of the isolated prop model. It is not apparent, even though the data indicates otherwise, that the wing and flaps have a favorable influence on the effectiveness in the C_{T_S} regime where the cyclic requirements are largest (higher C_{T_S} values), since it is not apparent that the increase in effectiveness recorded on the full span model is not merely reflecting the favorable change in prop normal force with cyclic previously noted in the hover mode and discussed on Page 62 in Reference 1.

The data slopes shown in Figure 41 for both horizontal tail-on and tail-off configurations were obtained from the curves of

94

slipstream pitching moment vs cyclic angle presented in Figure
42 and 43, respectively. These figures illustrate the uniform-
ity of the data over the conditions evaluated and the linearity
of the control through the range of cyclic angles tested.

The variation of pitching moment per degree of cyclic with
fuselage angle for a typical set of test runs, performed with
45° of wing tilt and at a nominal C_{T_S} of 0.81, is presented in
Figure 44 for both tail-on and tail-off cases. As indicated
for this example, and which is typical for the other conditions
tested, the effectiveness decreased at a moderate rate of
about 15% from the lowest to the highest shaft angle tested,
the important point being that no noticeable decrease occurred
as a result of wing stall. This decrease with shaft angle,
previously noted when analyzing the isolated prop data and
shown in Figure 41, is largely a ramification of the slipstream
notation system. All 1/12th scale test runs were performed
with a constant propeller RPM and tunnel speed or q, the result
being that propeller thrust and slipstream q (q_s) increased
with the propeller shaft angle, α_p. Therefore, a hub pitching
moment that was found to be virtually independent of shaft
angle in propeller notation will appear to decrease when trans-
cribed to the slipstream notation. Part of the decrease can be
identified with the small percentage change in lift accompany-
ing cyclic pitch inputs.

Both Figures 41 and 42 show some difference between the cyclic
pitch effectiveness measured with the tail on and off. The
data in Figure 41, for the descent case, indicates that the
tail exerts some favorable influence at thrust coefficient
values greater than 0.40.

SYM	i_W	SHAFT ANGLE, α_p
⊙	15°	20°
☺	√	√
⊡	30°	25°
⊡	√	√
△	45°	40°
◇	55°	55°

Open Symbols Denote Tail On
Solid Symbols Denote Tail Off

Isolated Prop (Hub Moment Only)
Fairings from BVWT 057
(Reference 2/$\theta_{.75}$=12° & 5000 RPM)

NOTES:
1. Model VRO68Q
2. Data from BVWT 067
3. Flaps at 60°
4. Full Span Slats
5. Mid c.g.
6. Average Collective Setting: 12°
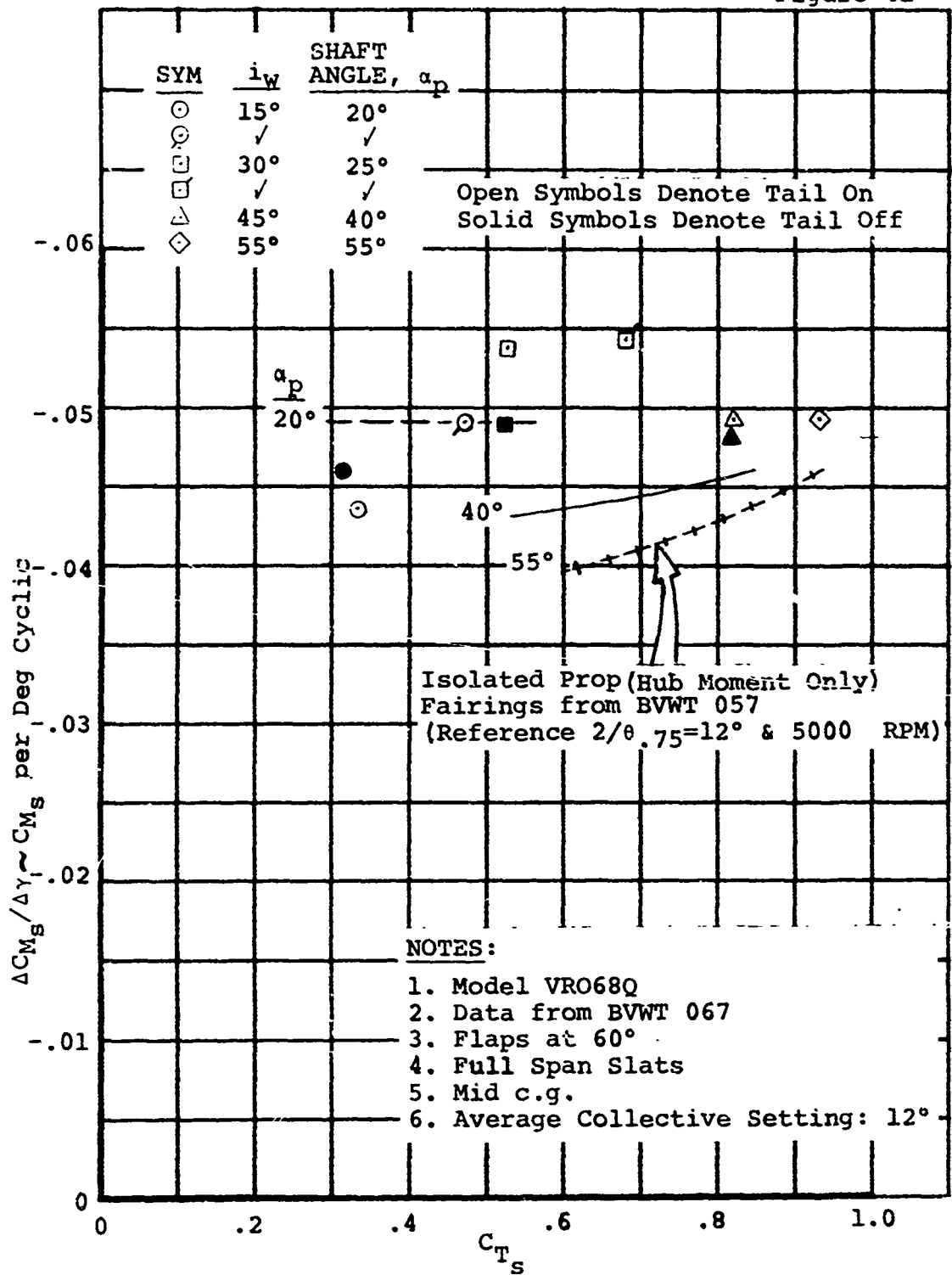
PITCHING MOMENT PER DEGREE CYCLIC
THROUGH TRANSITION
HIGH SHAFT ANGLES/DESCENT CASE

96

Figure 42

NOTES:

1. Model VR058Q
2. Data from BVWT 067
3. Flaps at 60°
4. Full Span Slats
5. Mid c.g.

SYM	i_w	SHAFT ANGLE, α_p	C_{T_S}
⊙	15°	20°	.335
⊙	√	√	.473
▢	30°	25°	.529
▱	√	√	.681
△	45°	40°	.821
◇	55°	55°	.934

α_p

55°

40°   25°

C_{M_S}

20°

HORIZONTAL TAIL ON

Cyclic Angle, $\gamma \sim$ Deg

LINEARITY OF CYCLIC PITCH CONTROL
THROUGH TRANSITION
HIGH SHAFT ANGLES/DESCENT CASE

SYM	i_w	SHAFT ANGLE, α_p	C_{T_s}
O	15°	20°	.318
□	30°	25°	.524
△	45°	40°	.819

NOTES:
1. Model VRO68Q
2. Data from BVWT 067
3. Flaps at 60°
4. Full Span Slats
5. Mid c.g.

HORIZONTAL TAIL OFF
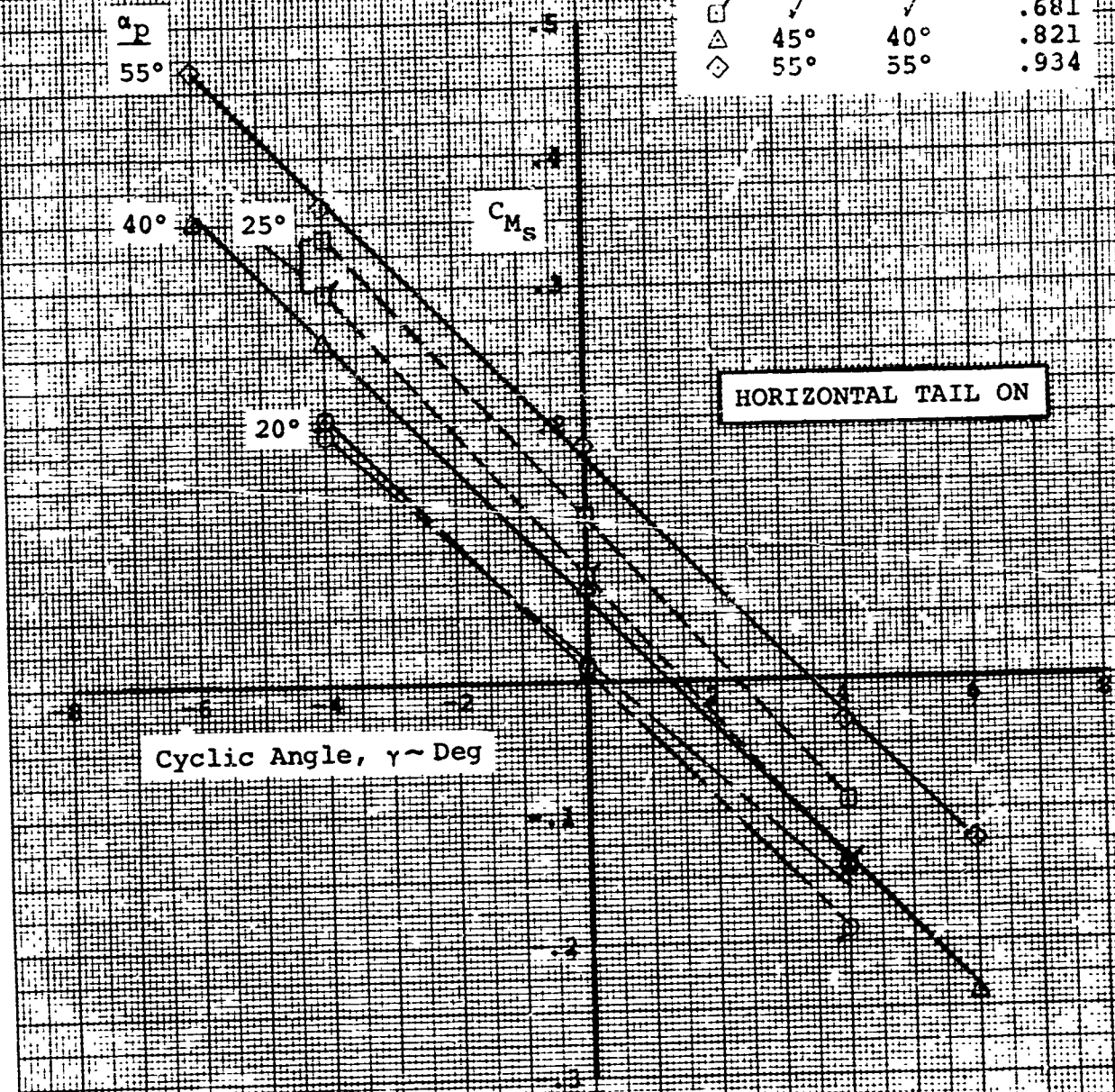
LINEARITY OF CYCLIC PITCH CONTROL
THROUGH TRANSITION
HIGH SHAFT ANGLES/DESCENT CASE

Figure 44

Fuselage Angle, $\alpha_F \sim$ Deg

VARIATION OF CYCLIC PITCH EFFECTIVENESS
WITH FUSELAGE ANGLE
$i_w = 45°$, $\delta_F = 60°$, NOM $C_{T_S} = .81$
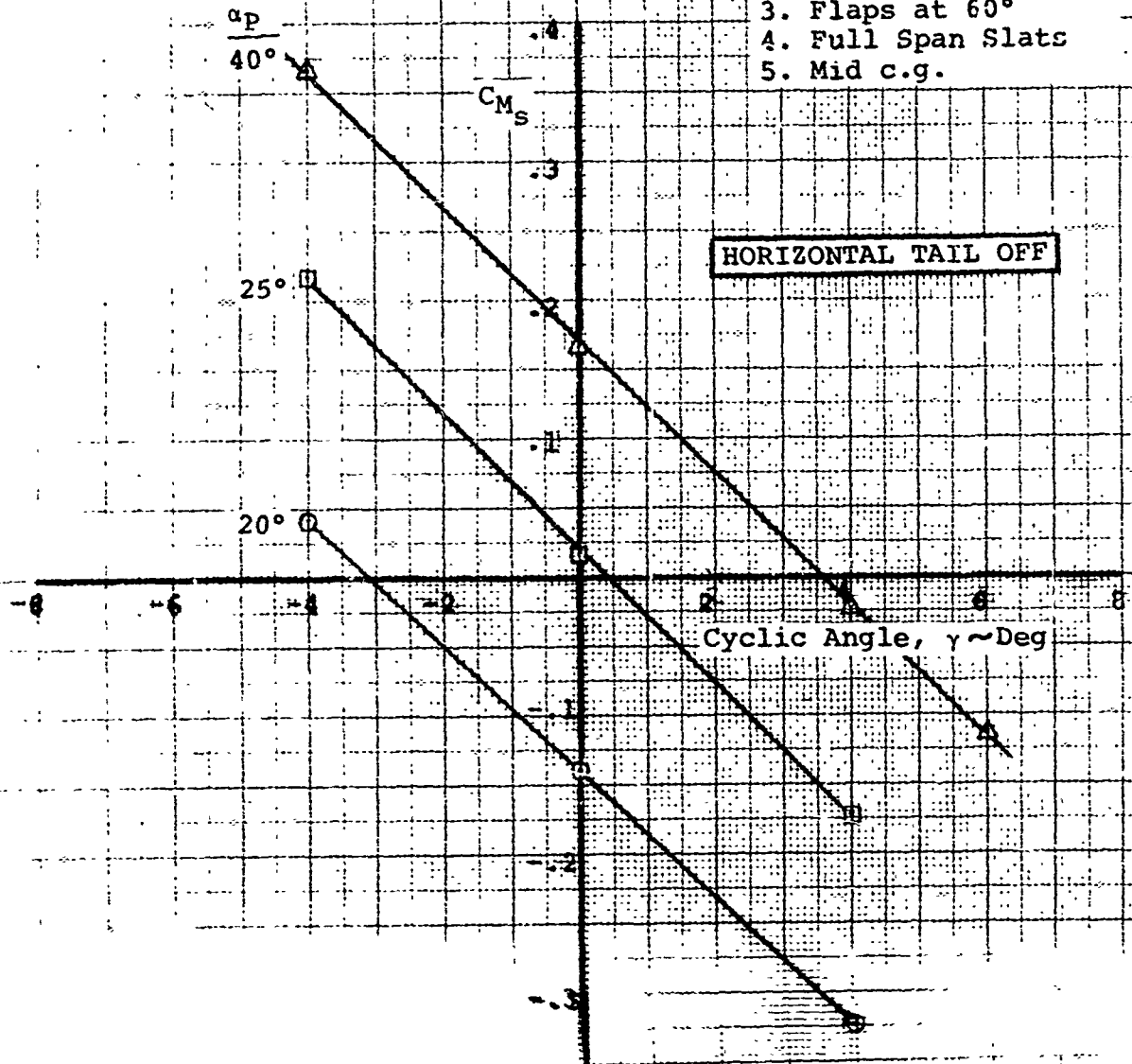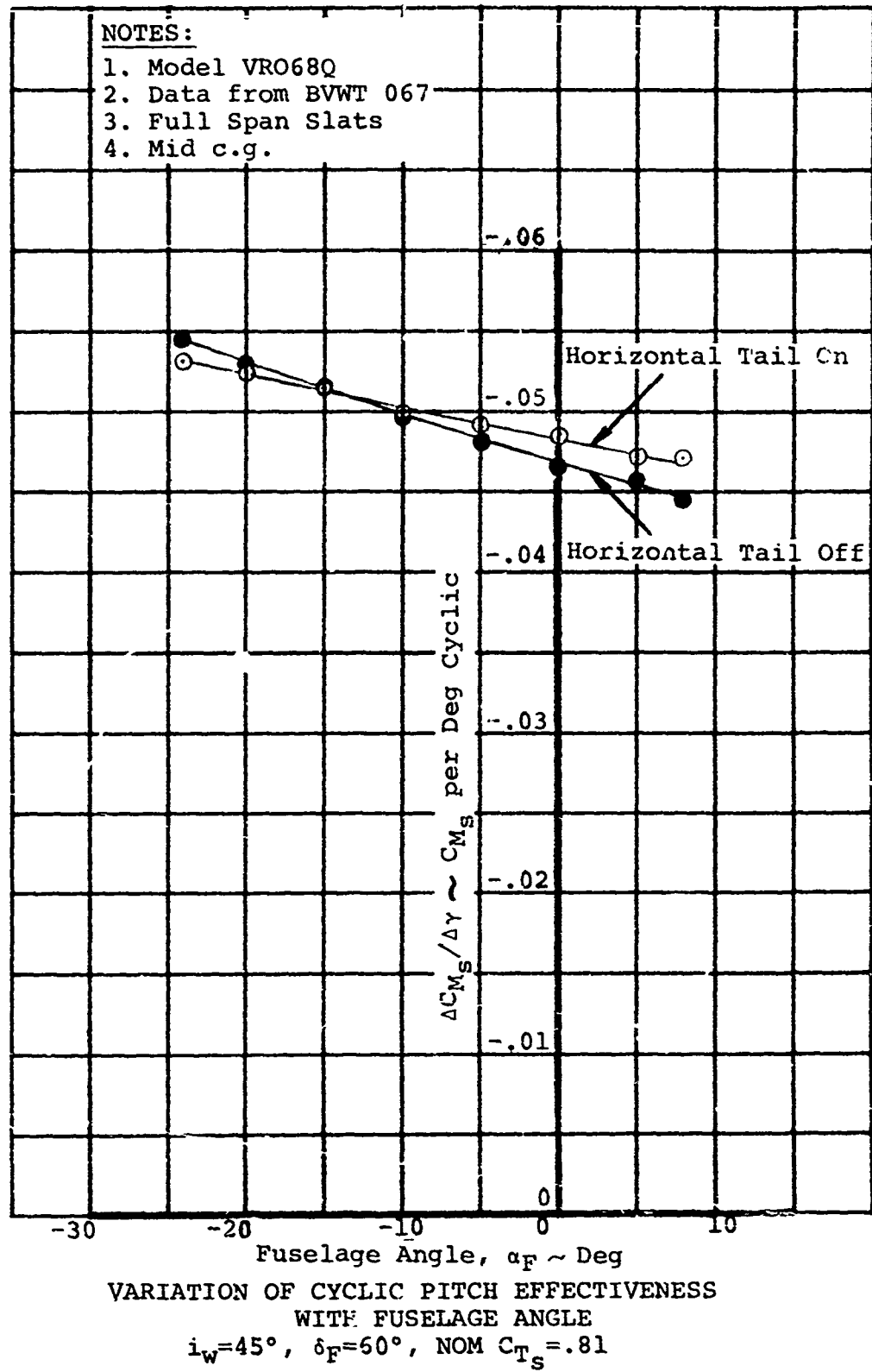
o.4.2 Effect of Cyclic Pitch on Longitudinal Stability

The effect of cyclic pitch on longitudinal stability is in-
herent in the test runs performed both horizontal tail on and
off for the purpose of establishing the cyclic pitch effec-
tiveness. Figure 45 presents the measured tail-off and tail-
on stability in the slipstream derivative form $C_{M_{S_\alpha}}$. The data
shown for selected combinations of wing tilt angle and slip-
stream thrust coefficient through transition with 60° of flap,
is applicable to the mid c.g. position described in Figure 40
which moves up and aft as the wing is tilted about the pivot
located 42.6% of the mean aerodynamic chord (MAC) aft of the
wing leading edge and 11.7% MAC below the wing chord plane.

Slopes shown in Figure 45 were measured from the pitching
moment plots (C_{M_S} vs α_F) presented later in this report section
and were extracted from the linear portion of the curves. The
C_{T_S} value designated for each data point is the average value
for the angle of attack range considered in choosing a slope.

Figure 45 presents two sets of data for the zero cyclic case.
The data denoted by solid lines was obtained with a propeller
RPM of 6800 during the Phase I test and the plotted points were
acquired at the cyclic hub operating speed of 5000 RPM. The
object of the dual set of data was to ascertain the quality of
the information acquired at a Reynolds number lower than 1.2
(10^6) achieved with a RPM of 6800. The effect of stability can
be noted to be small.

Figure 45 indicates that the application of cyclic pitch does
not have a marked effect on longitudinal stability with the
effect diminishing with increasing thrust coefficients or de-
creasing flight speeds. Positive cyclic angles can be seen to
increase the tail-off instability and negative cyclic decrease
the instability. Since the same incremental effects were
recorded both tail-on and tail-off, the changes in stability
probably reflect the combined influence of cyclic pitch on (1)
propeller hub moment and normal force when interpreted in the
slipstream notation and on (2) aircraft lift curve slope that in-
creases with negative cyclic and decreases with positive
cyclic.

Of concern during the testing was whether the thrustline off-
set occurring with cyclic pitch application would modify the
downwash gradient and thus produce substantial changes to
tail-on stability, especially at the lower range of thrust co-
efficients where the horizontal tail is particularly effective
for stability and control. The change in q at the tail plane
as a result of effectively raising or lowering the thrustline

100

with cyclic pitch would act in the opposite direction and tend
to provide for example, increased tail-on stability with posi-
tive cyclic angles, Apparently the high tail positioning re-
duces any influence of cyclic pitch on the downwash gradient
and q at the tail to a minimum with the only noticeable in-
fluence occurring at the lowest thrust coefficient tested and
15° of wing tilt. At this condition of 0.28 C_{T_S}, the incre-
mental change from tail-off to tail-on stability increased
with positive cyclic and decreased with negative cyclic, thus
indicating the q change at the tail to be the predominate
effect.

Figures 46 through 77 include the basic data from which the
cyclic pitch control capability and its effect on longitudinal
stability were established. Figures 46 through 66 and Figures
67 through 77 present horizontal tail-on and horizontal tail-
off data, respectively. The data, in slipstream notation, is
presented in sets of three plots, C_{M_S} vs. α_F (pitching moment
curves), C_{L_S} vs. C_{X_S} (force polars), and C_{L_S} vs. α_F (lift curves),
in the order of increasing wing tilt angle and slipstream
thrust coefficient, C_{T_S}. In these plots, CMCS, CLCS, and CXCS,
which represent wind axis data, should be read as C_{M_S}, C_{L_S} and
C_{X_S}, respectively. ALPHA signifies fuselage angle.

The following graphs contain series of runs with different
cyclic pitch settings, all conducted at a constant propeller
RPM or nominal C_{T_S} and a selected wing tilt/flap configuration.
Noted on each force polar presentation are lines corresponding
to 10° descent and 10° climb conditions. Interspersed between
the sets of data are plots of the variation of slipstream
thrust coefficient that occurred during the test runs.

Figure 63 presents typical full span model cyclic pitch control
data, in this case, data acquired with 55° of wing tilt at C_{T_S}
of 0.93 which represents a forward flight pat speed of
approximately 30 knots. The linearity of the change in pitch-
ing moment with cyclic angle and the small variation in effec-
tiveness with angle of attack is characterstic of the other
cyclic pitch control data that was obtained.

An examination of the individual pitching moment plots, for
example Figure 46 (15° wing tilt/60° flaps/.3 C_{T_S}), will reveal
that the pitching moment varies linearly with fuselage angle
of attack over the initial portion of the angle of attack range
tested. At a particular angle of attack for a given test
condition, the tail-on stability ($C_{M_{S_\alpha}}$) appears to increase.

101

This characteristic was identified in Section 6.4.4 of Reference 1 as being concurrent with flow separation on top of the fuselage that in turn decreased the downwash gradient prevailing at the horizontal tail. Reference 1 also discussed the change in character of the flow aft of the tilted wing and lack of separation at combinations of high wing tilt angles (45° and above) and high thrust coefficients (.92 C_{T_S}) and the resulting linearity of the pitching moment curves over the angle of attack range evaluated. See Figure 64.

As stated previously, a comparison of the pitching moment data obtained during this test (BVWT 067) with a propeller speed of 5000 RPM and the data acquired during the Phase I test (BVWT 061) with a propeller speed of 6800 RPM showed little change in the measured aircraft stability. This comparison also showed a positive shift in pitching moment between the Phase II data and the Phase I data that amounted to 0.06ΔC_{M_S} at 0.3 C_{T_S} and 15° of wing tilt and a similar value at 0.81 C_{T_S} and 45° of wing tilt. The reason for this shift has not been firmly established at this time, but a source of a portion of the shift could have arisen as a result of a lower flap effectiveness ensuing from the lower Reynolds number of the Phase II test.

Figure 45

EFFECT OF CYCLIC PITCH ON
LONGITUDINAL STABILITY

Figure 46

SHEET 104
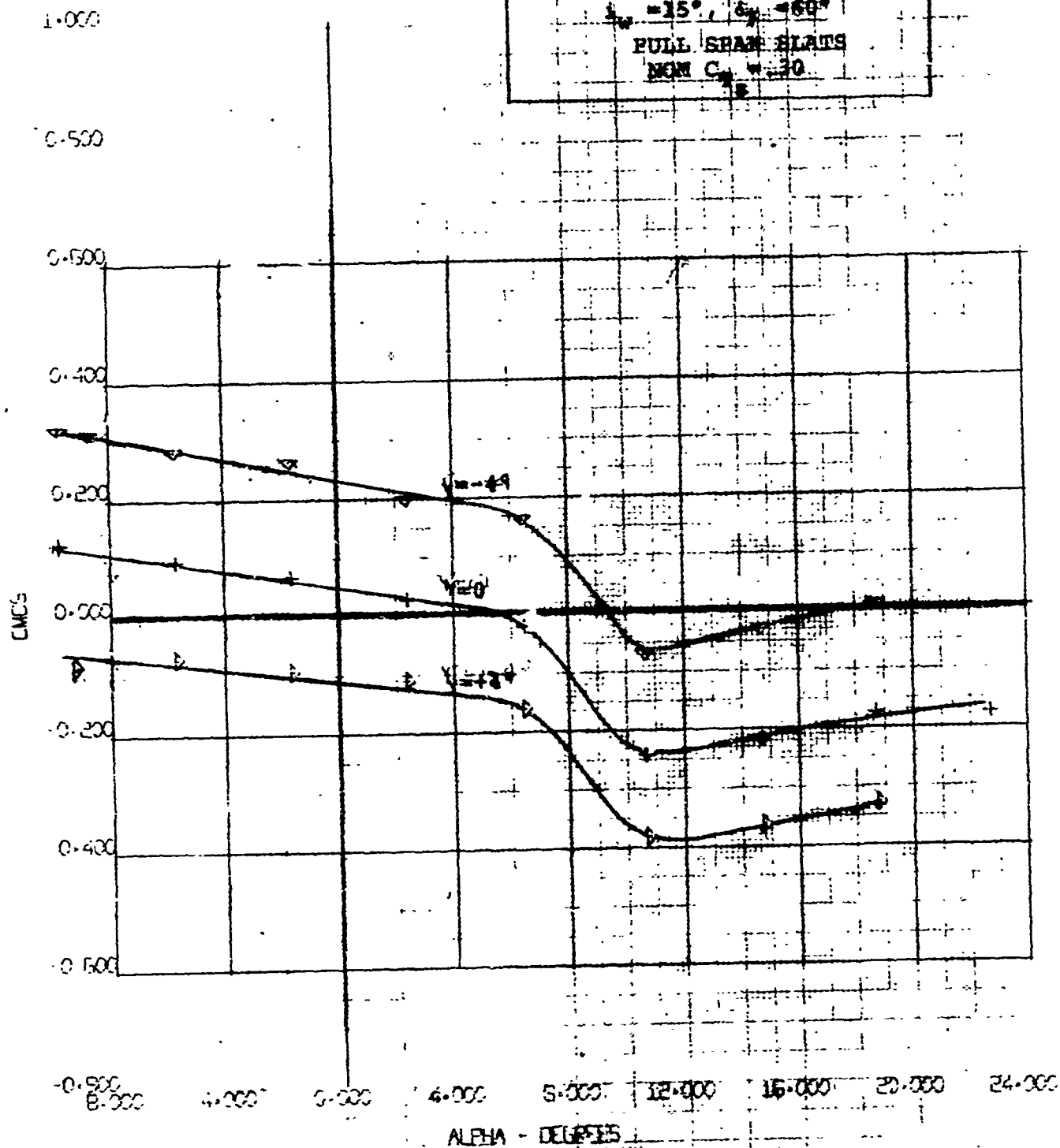
FOUR-PROP TILT WING
MODEL VRC58G (FULL SPAN)
CLCS VS. CXCS

BVWT 67
12/8/70

NOT REPRODUCIBLE

SHEET 105

RUN	SYM	↗	q	Y
31	+	+5°	12.1	0
33	▷	↓	↓	+4°
39	▽			-1°

CYCLIC PITCH CONTROL/TAIL ON
$i_w = 15°$, $\delta_p = 60°$
FULL SPAN SLATS
NOM $C_m = .30$

FOUR-PROP TILT WING
MODEL VROEBG (FULL SPAN)
CLCS VS. ALPHA
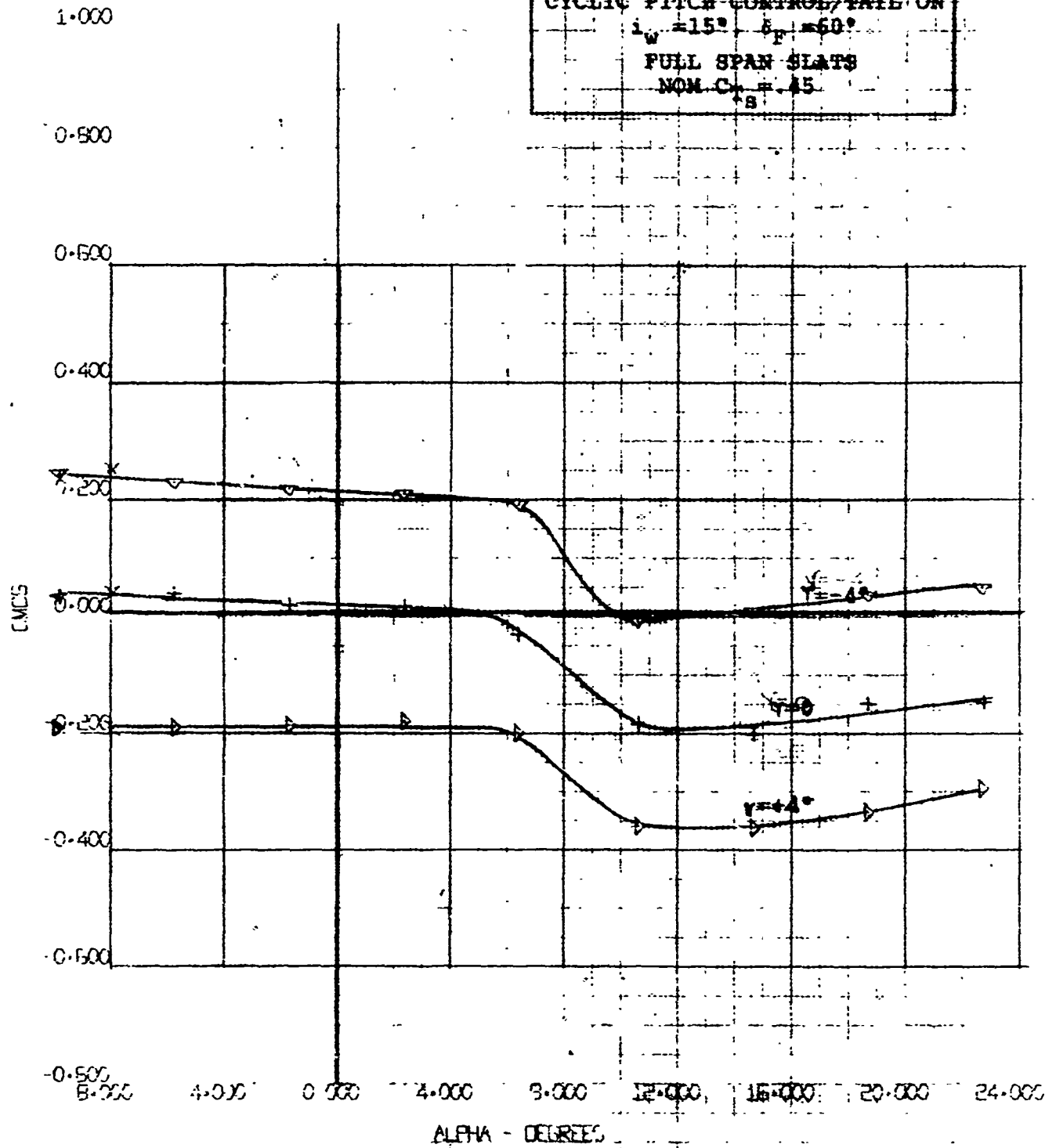
BVWT
67

12/ 8.70

SHEET 106

SHEET 107

RUN	SYM	i	C	Y
30	+	+5°	9.1	0
32	▷			+4°
38	▽			-4°

CYCLIC PITCH CONTROL/TAIL ON
$i_w = 15°$, $\delta_f = 60°$
FULL SPAN SLATS
NOM $C_{T_s} = .45$
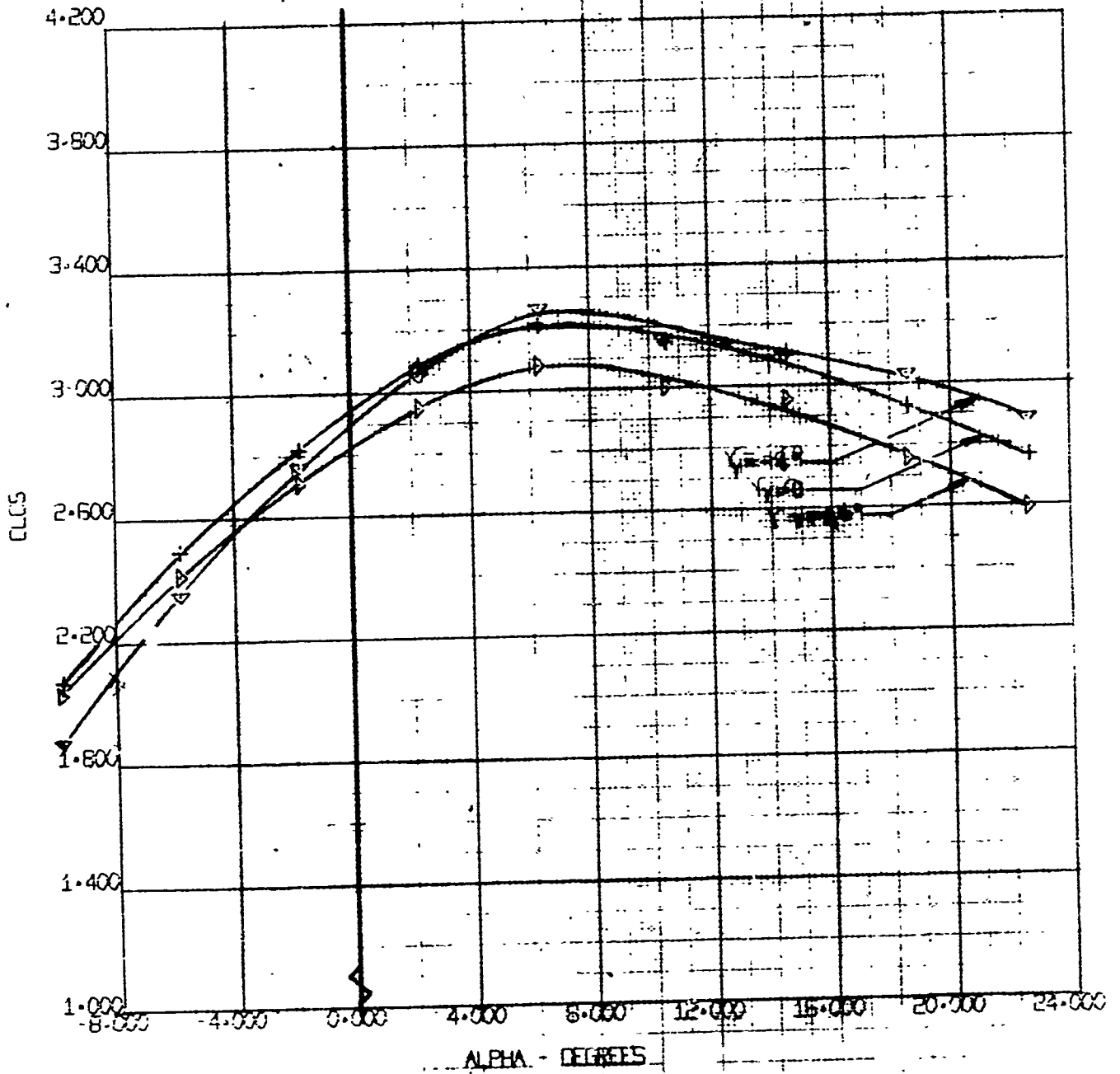
FOUR-PROP TILT WING	BVWT
MODEL VR068O(FULL SPAN)	67
CLCS VS. CXCS	12/6/70

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
CLCS VS. ALPHA

Figure 51

Figure 52

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE
$i_W = 15°$, $\delta_F = 60°$
5000 RPM

SHEET 110

Figure 53. FOUR-PROP TILT WING MODEL VRC480 (FULL SPAN) CMC5 VS. ALPHA

Figure 54

FOUR-PROP TILT WING
MODEL VZ060(FULL SPAN)
CLCS VS. CXCS

BVWT
67

12/10/70

SHEET 112

Figure 55
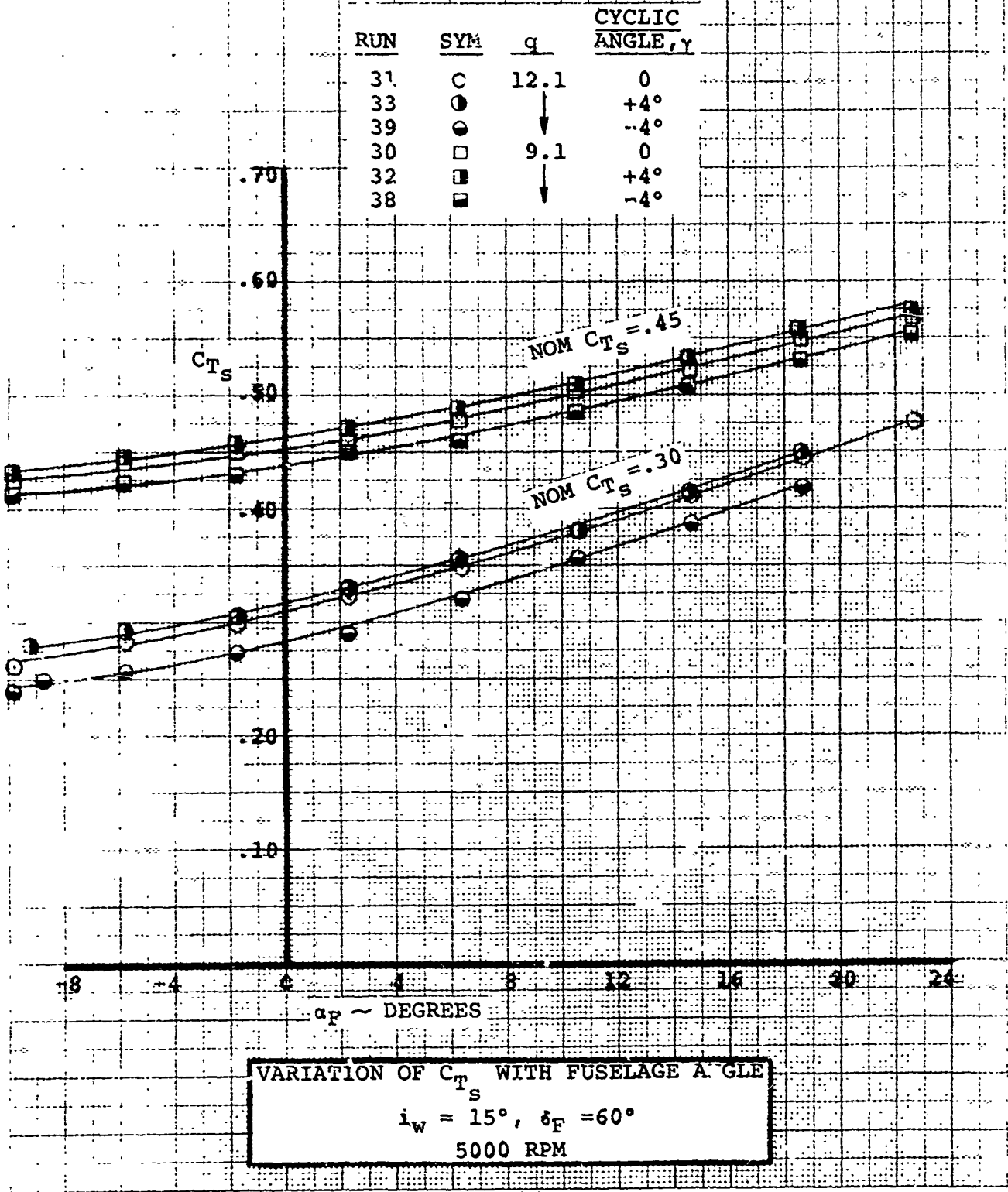
CYCLIC PITCH CONTROL/TAIL ON
$i_w = 80°$, $\delta_T = 60°$

FULL SPAN FLAPS
NOM $C_{T_s} = .55$

RUN	SYM	δ	α	γ
50	+	+15°	8.2	0°
35&61	▷ & ▷			+4°
37&70	▽ & ▽			-4°

ALPHA - DEGREES

CLCS

NOT REPRODUCIBLE

SHEET 113

FOUR-PROP TILT WING
MODEL VROGRO (FULL SPAN)
CLCS VS. ALPHA

BVWT
67

12/10/70

NUMBER D170-10039-1
REV. LTR.

CYCLIC PITCH CONTROL/TAIL CLJ
$i_w = 30°$, $i_a = 60°$
FULL SPAN SLATS
NOM $C_T = 79$

RUN	SYM	⊄	q	Y
51	+	+15°	5.3	0
34&60	▷≤△	↓	↓	+4°
36&69	▽&◁	↓	↓	-4°

1·000

0·500

FOUR-PROP TILT WING
MODEL VR0880 (FULL SPAN)
CMCS VS· ALPHA

BVWT 67
12/10/70

RUN	SYM	α	q	γ
51	+	+15°	5.3	0
34&60	▷ & ▷	↓	↓	+4°
36&69	▽ & ✕	↓	↓	-4°

CYCLIC PITCH CONTROL/TAIL ON
$i_W = 30°$, $\delta_R = 63°$
FULL SPAN SLATS
NOM $C_{T_S} = .70$

±9° Descent
Angle

±9° Climb
Angle

CLLS

CXCS

FOUR-PROP TILT WING	BVWT
MODEL VSC360(FULL SPAN)	67
CLLS VS. CXCS	12/10/70

NOT REPRODUCIBLE

SHEET 115

CYCLIC PITCH CONTROL/TAIL ON
$i_w = 39°$, $\delta_f = 60°$
FULL SPAN SLATS
NOM $C_T = .70$

RUN	SYM	i_w	δ_f	γ
51	+	+15°	8.3	0
34&60	▷&▷			+4°
36&69	▽&▽			-4°

NUMBER D170-16039-1
REV. LTR.
FIGURE 58

FOUR-PROP TILT WING
MODEL VROBO (FULL SPAN)
C_L VS. ALPHA

BWT
67
12/10/70

ALPHA - DEGREES

SHEET 116
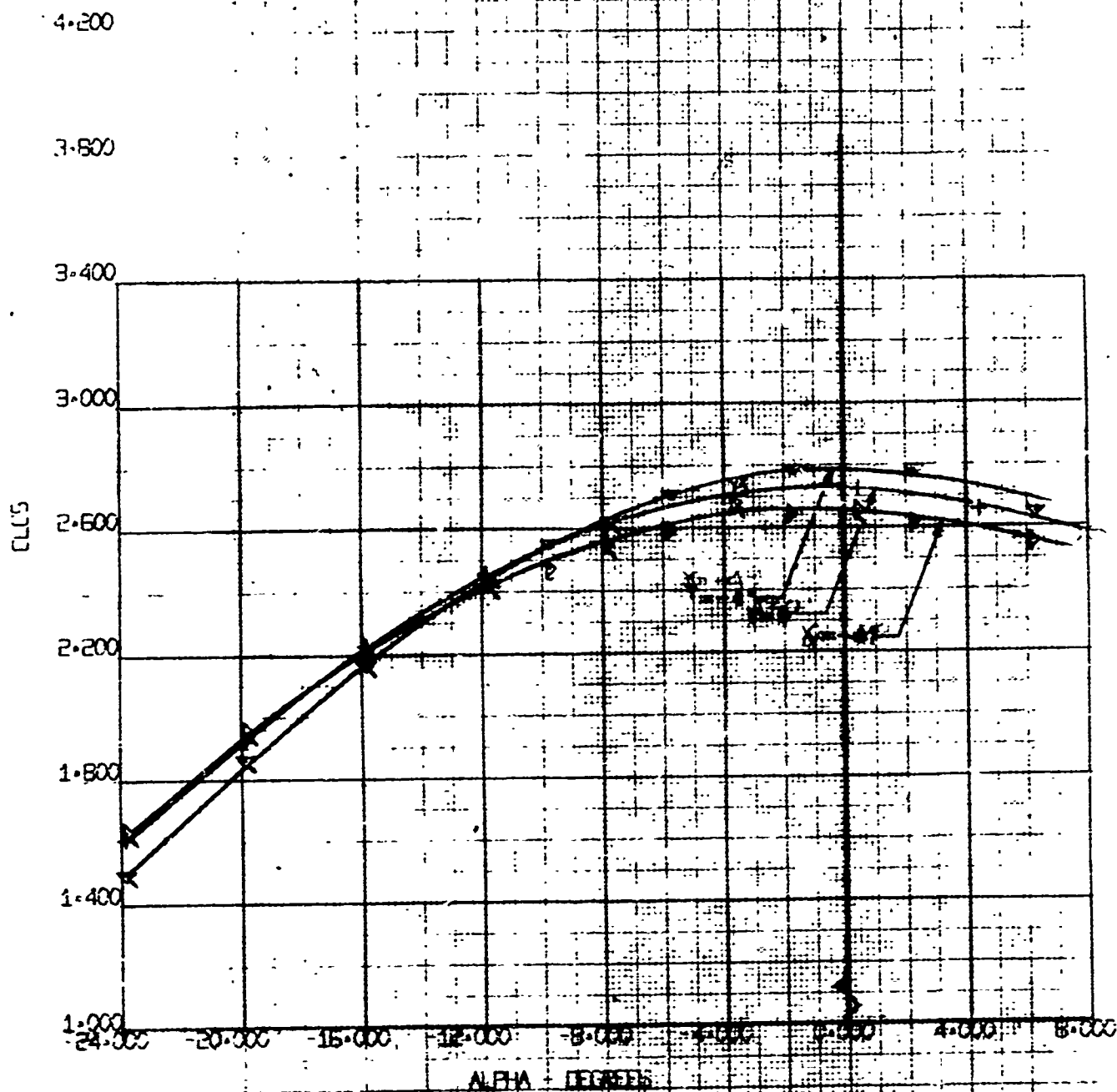
Figure 59

NOTES:
1. ~~Note. BW,BW~~
2. Data from BVWT 067
3. Full Span Slats
4. Tail On ~ $\measuredangle$ =+15°

RUN	SYM	ϵ	CYCLIC ANGLE, γ
50	○	8.2	0
35	◑		+4°
61	◐		+4°
37	◓		-4°
70	◒		-4°
51	□	5.3	0
34	◨		+4°
60	◧		+4°
36	⬓		-4°
39	⬒		-4°

C_{T_S}

1.00

.90

.80

NOM C_{T_S}=.70

.70

NOM C_{T_S}=.55

.60

.55

.50

.40

-24 -20 -16 -12 -8 -4 0 4 8

α_F ~ DEGREES

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE

i_W = 30°, δ_F =60°

5000 RPM

NOT REPRODUCIBLE   SHEET 117

Four-Prop Tilt Wing / Model VRC380 (full span) / CMCS vs. Alpha — BVWT 67, 12/10/70

RUN	SYM	⌐	q	RPM	γ
52	+	+35°	3.33	5000	0
54	▷		3.33	5000	+4°
171	◁		2.70	4500	+6°
65	▽		3.33	5000	-4°
169A	△		2.70	4500	-6°

CYCLIC PITCH CONTROL/TAIL ON
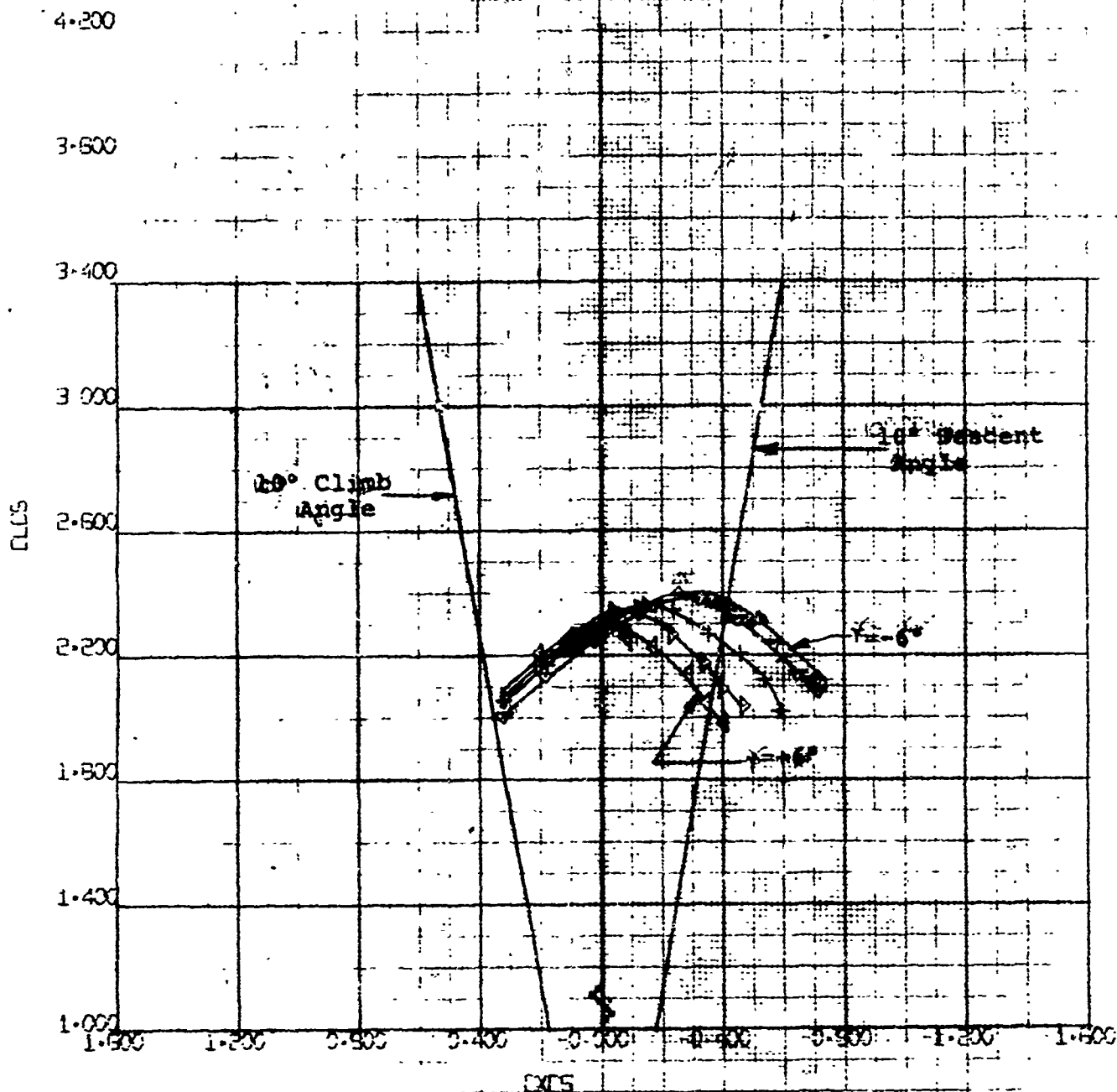
FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VR05B0 (FULL SPAN)
CLCS VS. CXCS

BVWT
67

12/10/70

NOT REPRODUCIBLE   SHEET 119

RUN	SYM	∡	q	RPM	Y
52	+	+35°	3.33	5000	0
54	▷		3.33	5000	+4°
171	◁		2.70	4500	+6°
65	✕		3.33	5000	-4°
169A	△		2.70	4500	-6°

NUMBER   D170-10039-1
REV. LTR.

Figure 62

CYCLIC PITCH CONTROL/TAIL ON
i_w =45°, δ_F =60°
FULL SPAN SLATS
C_{T_s} = .81

FOUR-PROP TILT WING
MODEL M160C (FULL SPAN)
CLS VS. ALPHA

BVWT
67

12/10/70

SHEET 120

Figure 63

CYCLIC PITCH CONTROL/TAIL ON
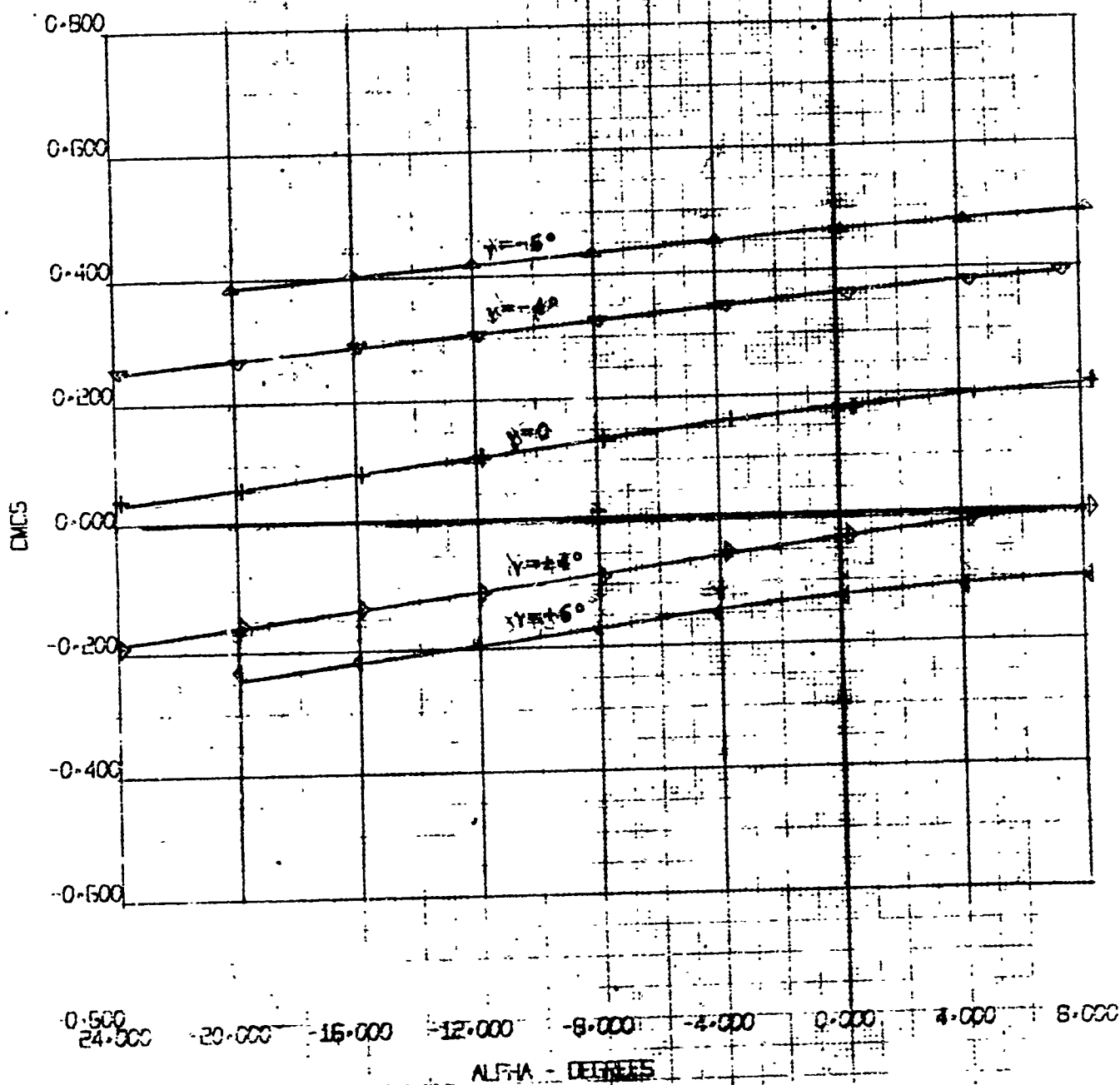
FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VRO58G (FULL SPAN)
CMC5 VS. ALPHA

NUMBER D170-10039-1
REV. LTR.

BVWT
67
12/10/70

SHEET 121

Figure 64

FOUR-PROP TILT WING
MODEL VROBEC (FULL SPAN)
CLCS VS. CXCS

CYCLIC PITCH CONTROL/TAIL ON
$i_w=55°$, $\delta_F=60°$
FR. & SPAN SLATS
NAC C_T = .93

RUN	SYM	∢	q	RPM	Y
53	+	+35°	1.16	5000	0
55	▷		1.16	5000	+4°
170	◁		1.00	4500	+6°
66	▽		1.16	5000	-4°
169	△		1.00	4500	-6°

FOUR-PROP TILT WING
MODEL VROBBO(FULL SPAN)
CLCS VS. ALPHA

BVWT 67

12/10/70

NOT REPRODUCIBLE

SHEET 123

Figure 66

NOTES:

1. Model VR068Q
2. Data from B/W 067
3. Full Span Slats
4. Tail On ~ ∂ =+35°

RUN	SYM	q	RPM	i_W	CYCLIC ANGLE ~ γ
52	○	3.33	5000	45°	0
54	◑	3.33	5000		+4°
171	◐	2.70	4500		+6°
65	◡	3.33	5000		-4°
169A	⊖	2.70	4500	↓	-6°
53	□	1.16	5000	55°	0
55	◨	1.16	5000		+4°
170	◧	1.00	4500		+6°
66	⊟	1.16	5000		-4°
169	⊟	1.00	4500	↓	-6°

NOM C_{T_S} =.93

NOM C_{T_S} =.81

C_{T_S}

α_F ~ DEGREES

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE

i_W = 45° & 55°, δ_F = 60°

5000 & 4500 RPM

Figure 67

CYCLIC PITCH CONTROL/TAIL OFF
FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VR0660 (FULL SPAN)
CMC5 VS. ALPHA

NUMBER D170-10039-1
REV. LTR.

BVWT 67
12/ 8/70

SHEET 125

FOUR-PROP TILT WING
MODEL VZ0880 (FULL SPAN)
CLCS VS. CXCS

BVWT 67

12/8/70

SHEET 126

RUN SYM

42 ▽
43 ▷
47 +

CYCLIC PITCH CONTROL/TAIL OFF

FULL SPAN GLAPS

4·200

3·800

3·400

3·000

CLCS

2·600

2·200

1·800

1·400

1·000

-8·000 -4·000 0·000 4·000 8·000 12·000 15·000 20·000 24·000

ALPHA - DEGREES

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
CLCS VS· ALPHA

BVWT
67

12/ 8/70

SHEET 127

NOTES:

1. Model VRO68Q
2. Data from BVWT 067
3. Full Span Slats
4. Tail Off

RUN	SYM	q	CYCLIC ANGLE, γ
47	⊙	12.1	0
43	◑	↓	+4°
42	◕		−4°

C_{T_S}

NOM $C_{T_S} = .30$

$\alpha_F \sim$ DEGREES

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE
$i_w = 15°$, $\delta_F = 60°$
5000 RPM

SHEET 128

CYCLIC PITCH CONTROL/TAIL OFF

FULL SPAN SLATS
NOM C_{μ} = .55

RUN	SYM	Δ	q	γ
48	+	---	8.2	0°
58	▷	---	↓	+4°
67	▽	---	↓	-4°

CMC5

ALPHA - DEGREES

FOUR-PROP TILT WING	BVWT
MODEL VRC580 (FULL SPAN)	67
CMC5 VS. ALPHA	12/8/70

SHEET 129

NOT REPRODUCIBLE

CYCLIC PITCH CONTROL TAIL OFF
i_w = -82°, i_F = 60°
FULL SPAN SLATS
NOM C_μ = .55

RUN	SYM	$\triangle$	q	γ
48	+	---	8.2	0°
58	▷	---		+4°
67	▽	---		-4°

4·200

3·500

3·400

3·000

CLLS

2·600

2·200

1·800

1·400

1·000

1·800   1·200   0·800   0·400   0·000   0·400   0·800   1·200   1·600

DXCS

16° Descent Angle

16° Climb Angle

FOUR-PROP TILT WING
MODEL VZ086 (FULL SPAN)
CLCS VS. DXCS

BVNT 67

12/8/70

SHEET 130

Figure 73. FOUR-PROP TILT WING MODEL VROSS (FULL SPAN) CLCS VS. ALPHA

CYCLIC PITCH CONTROL/TAIL OFF
i_w = 60°, δ_F = 60°
FULL SPAN SLATS
NOM C_μ = 5

RUN	SYM		q	y_c
48	+		8.2	0°
58	Δ	---		+4°
67	▽	---		-4°

SHEET 131
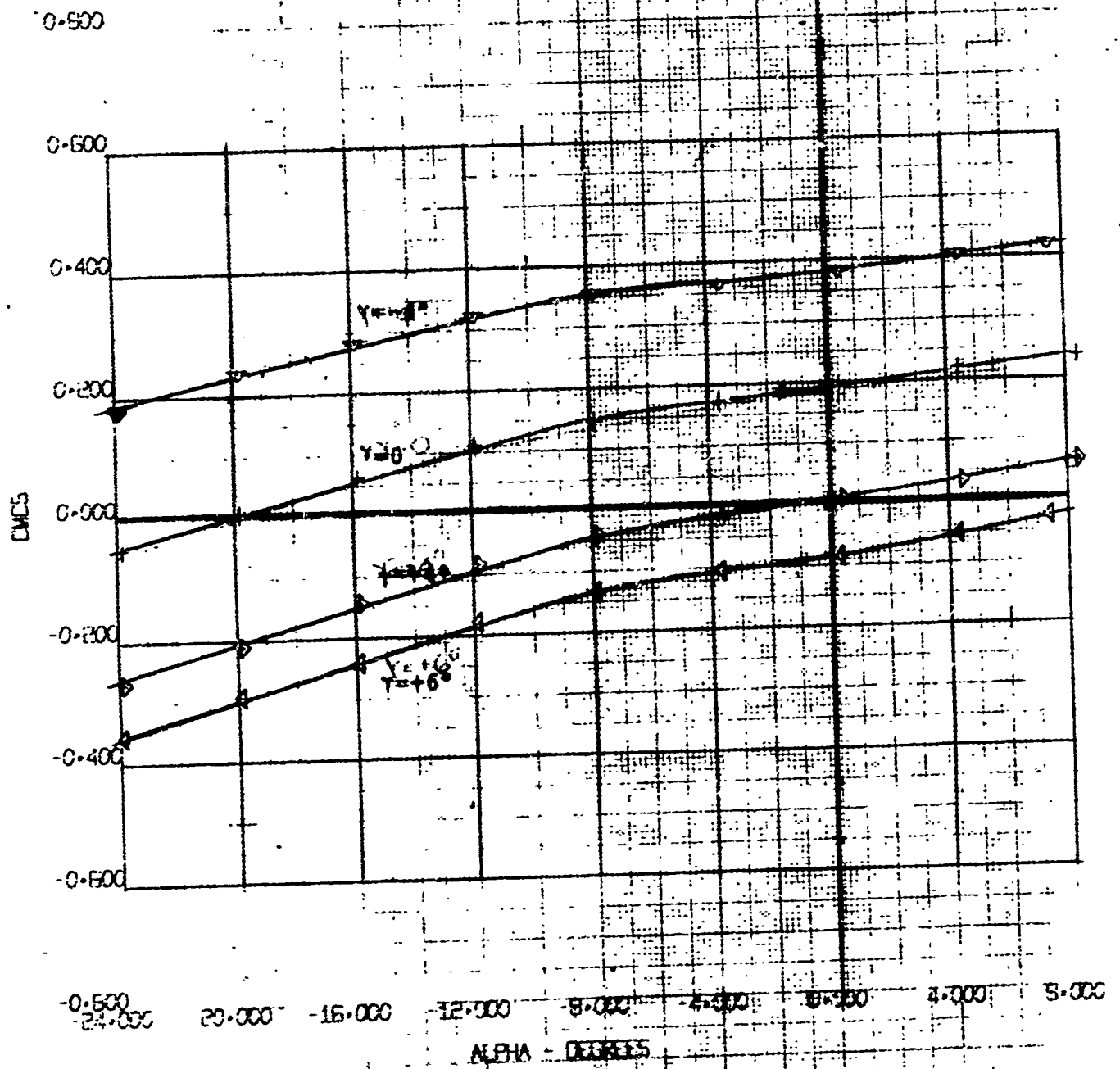
RUN	SYM	Δ	α	Y
49	+	---	3.33	0°
59	⊕	---		+4°
68	✳	---		-4°
71	⊿	---		+6°

CYCLIC PITCH CONTROL/TAIL OFF
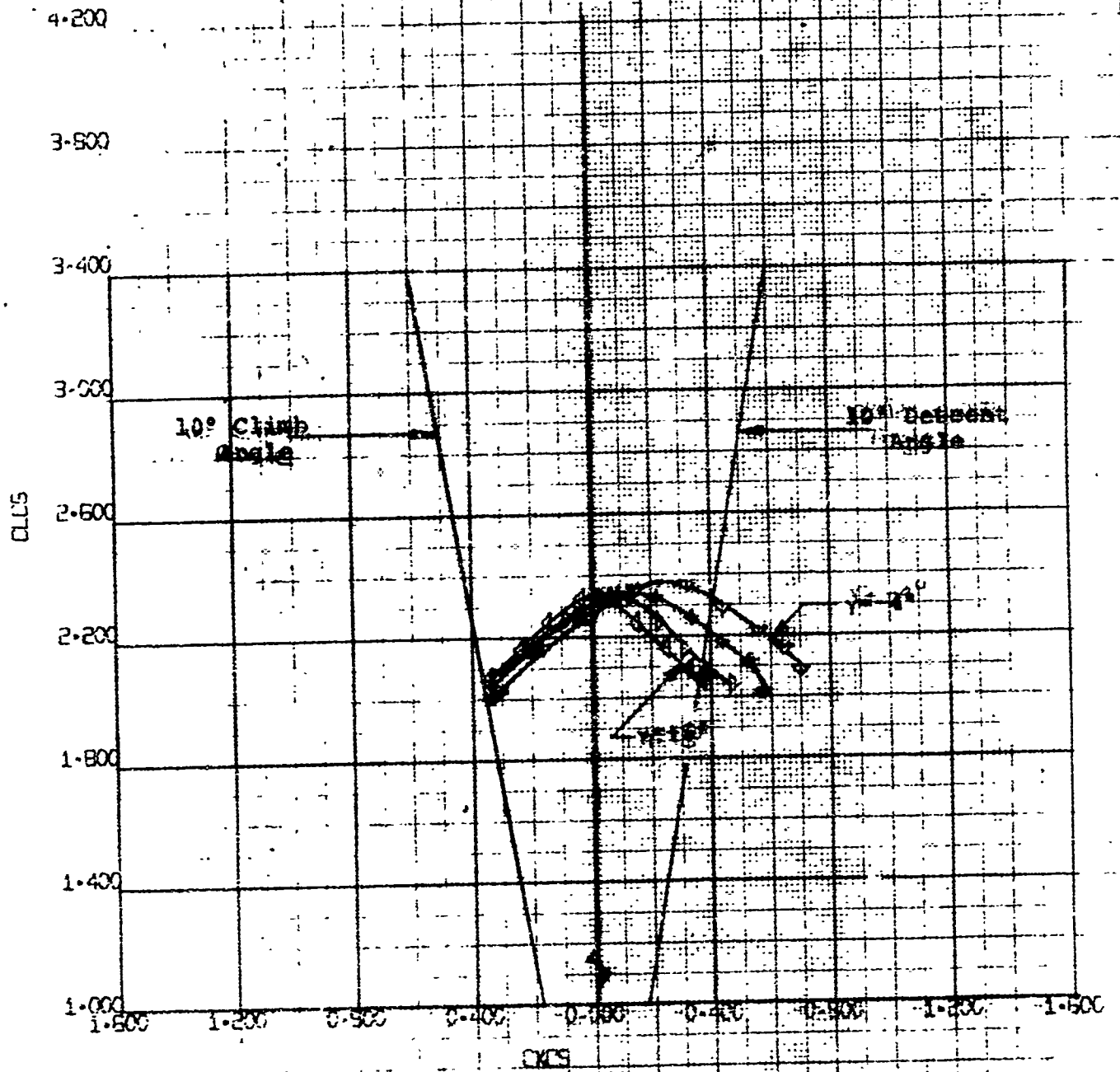
i_w = 45°, i_f = 60°

FULL SPAN SLATS

NOM C_T = .81

CMC5

ALPHA - DEGREES

FOUR PROP TILT WING
MODEL VROSO (FULL SPAN)
CMC5 VS. ALPHA

BVWT 67

12/10/70

SHEET 132

Figure 75. CYCLIC PITCH CONTROL/TAIL OFF — FOUR-PROP TILT WING, MODEL VROG80 (FULL SPAN), CLCS VS. CXCS, BVWT 67, 12/10/70, SHEET 133

FOUR-PROP TILT WING
MODEL VR0606 (FULL SPAN)
CLCS VS. ALPHA

BVWT
67

12/10/70

NOT REPRODUCIBLE
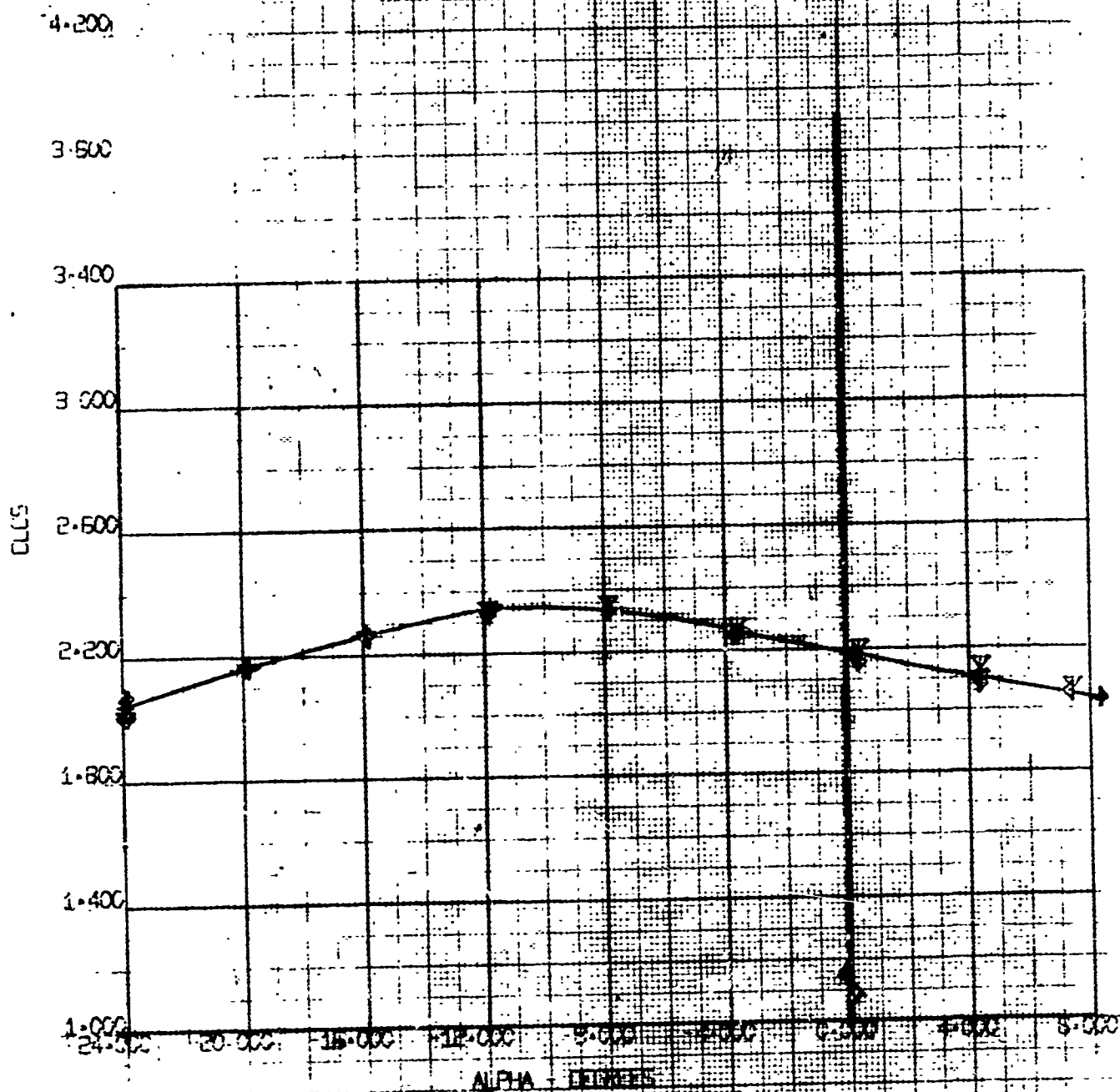
Figure 27

NOTES:
1. Model VRO680
2. Data from BVWT 067
3. Full Span Slats
4. Tail Off

RUN	SYM	q	CYCLIC ANGLE, γ	i_w
48	⊙	8.2	0°	30°
58	◑	↓	+4°	↓
67	◓		-4°	
49	▣	3.33	0°	45°
59	◧	↓	+4°	↓
71	◪		+6°	
68	▨		-4°	

NOM C_{T_S}=.81

C_{T_S}

NOM C_{T_S}=.55

α_F—DEGREES

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE
i_w=30°, & 45°, δ_F=60°
5000 RPM

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P.O. 340R-MP DIETZGEN GRAPH PAPER
MILLIMETER
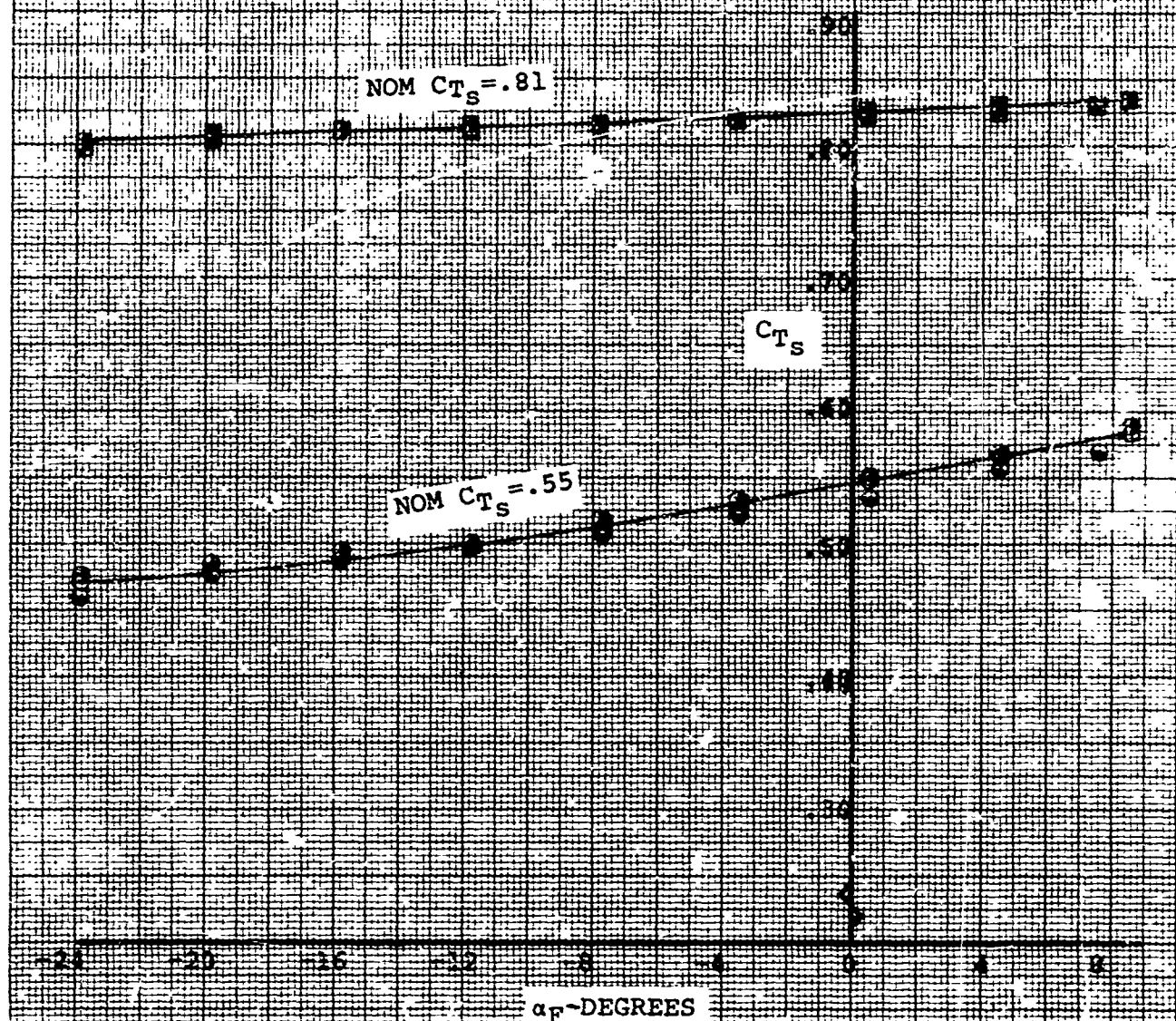
6.4.3 Horizontal Tail Effectiveness with Cyclic Pitch Inputs

Results of the investigation of horizontal tail effectiveness when longitudinal control is provided by coupled stabilizer and cyclic pitch control are presented in Figure 78. The test data in this figure is compared with the stabilizer effectiveness, in terms of $\Delta C_{M_S}/\Delta \lambda$ (increment of slipstream pitching moment per degree of stabilizer angle) obtained during the more extension investigation with zero cyclic and 6800 propeller RPM. The reduction in propeller speed to 5000 RPM for cyclic pitch operation, which decreased the slipstream q by almost 50%, did not have a significant effect on the measured tail power in the linear range over which the dat was interpreted.

Figure 78 shows that the application of cyclic pitch has only a small influence on the basic tail effectiveness in transition with the greatest effect occurring at .27 C_{T_S} with 15° of wing tilt. Positive cyclic can be noted to increase the tail power. This result is reasonable, since positive cyclic effectively raises the thrustline and thus should increase the q acting on the tail. Tilting the wing to a higher angle will lower the slipstream wake with respect to the tail chord plane and should diminish the effect. It can be assumed that cyclic pitch would have a larger effect on stabilizer effectiveness if the horizontal tail was located in a lower position.

Figures 79 through 93 present the basic three component data in sets of three plots: pitching moment curve, force polar, and lift curve for a common wing tilt angle, nominal C_{T_S} , and cyclic angle input. See Page 101 for a definition of machine symbols. Each set presents a series of runs with different tail incidences relative to a fuselage waterline. The corresponding tail-off run has also been shown, when available.

In Figure 85, for example, the progressive reduction in tail-on pitching moment at more negative fuselage angles and the convergence of the various stabilizer angle curves to a constant incremental difference in pitching moment from the tail-off curve, reflects the normal stall of a horizontal tail and thus the maximum pitching moment capability from the tail. Since the tail is mounted high on the fin and is essentially outside the influence of the propeller slipstream at most wing tilt angles, the maximum trim capability of the stabilizer will vary almost linearly with C_{T_S}, the pattern established for the horizontal tail effectiveness ($\Delta C_{M_S}/\Delta \lambda$). See Page 118 of Reference 1. The low Reynolds number acting on the model tail should

significantly decrease its maximum lift coefficient and result-
ing trim capability as compared to a full scale tail.
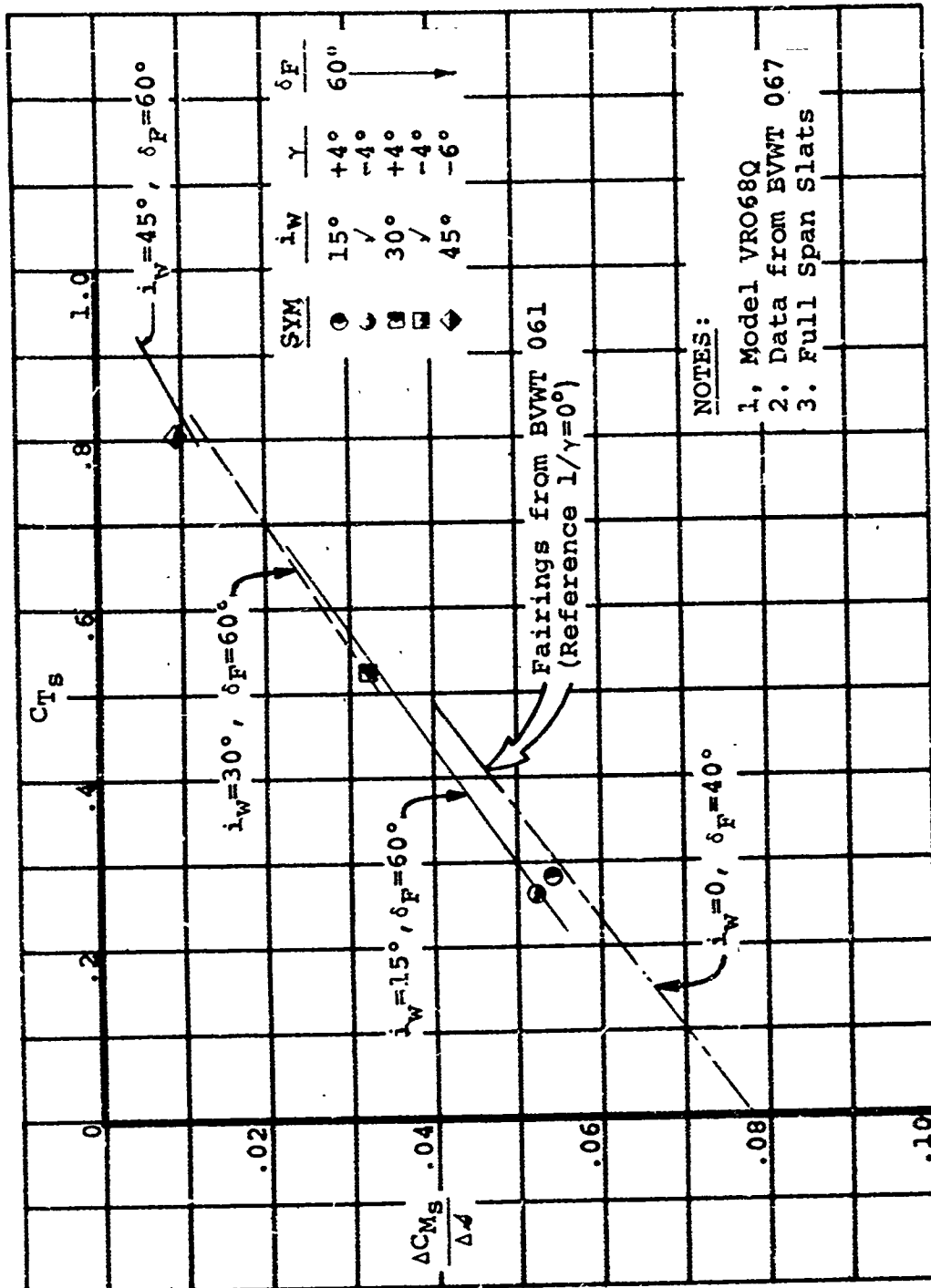
EFFECT OF CYCLIC PITCH ON
HORIZONTAL TAIL EFFECTIVENESS

STABILIZER CONTROL DATA PUBLIC

FULL SPAN SLATS
NOW $C_T = .30$

Figure 79

NUMBER D170F 18039-1
REV LTR-

RUN	SYM	i_t	q	i_T
43	+	---	12.1	+4°
45	◁	0°		
46	▽	+10°		
111	△	+5°		
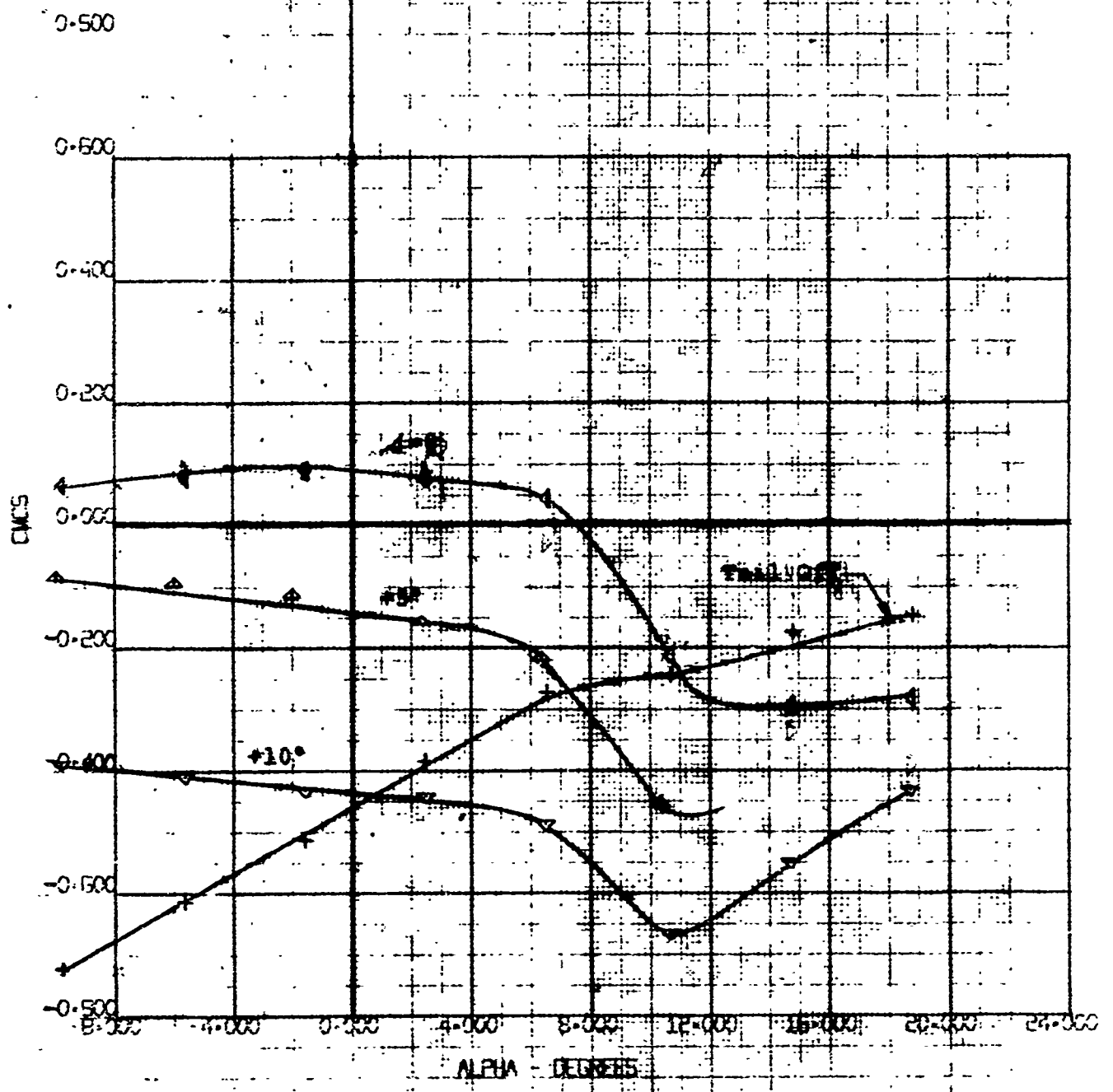
FOUR-PROP TILT WING	BWNT
MODEL VR080 (FULL SPAN)	67
CMCS VS. ALPHA	12/ 8/70

SHEET 139

FOUR-PROP TILT WING
MODEL VROGBC (FULL SPAN)
CLCS VS. CXCS

BVWT 67

12/ 8/70

NUMBER D170-10003-1
REV. LTR. Figure 81

RUN	SYM	4	q	Y
43	+	-4°	12.1	▲4°
45	◁	0°		
46	▽	+10°		

STABILIZER CONTROL WITH CYCLIC
$i_w = 15°$, $\delta_F = 40°$
FULL SPAN SLATS
NOM $C_T = 38$

FOUR-PROP TILT WING
MODEL VRC580 (FULL SPAN)
CLCS VS. ALPHA

BVWT 67
12/6.70

NOT REPRODUCIBLE   SHEET 141

FOUR-PROB TILT WING
MODEL VROBO (FULL SPAN)
CMCS VS. ALPHA

SHEET 142

SHEET 143
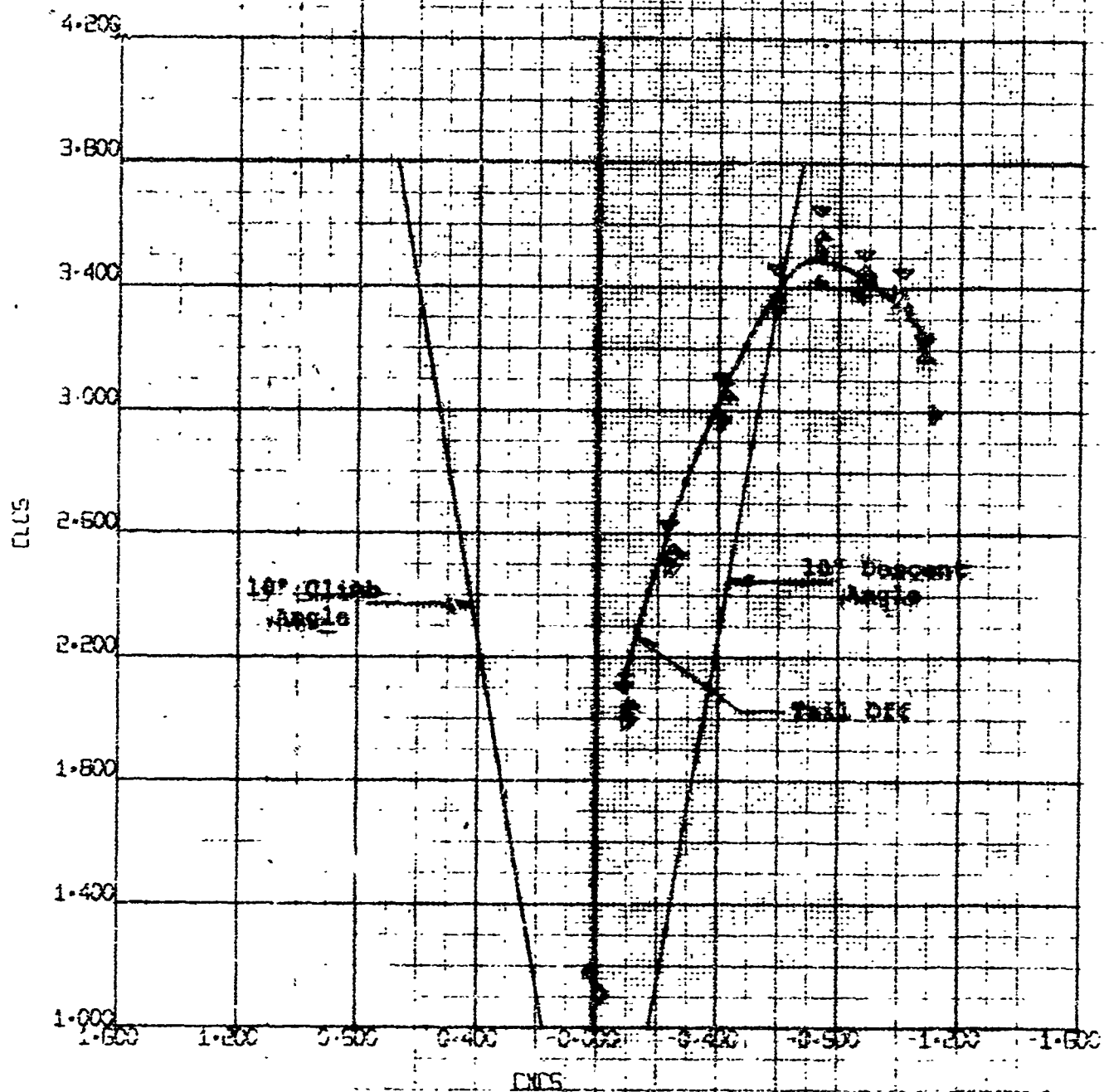
Figure: Four-prop tilt wing, Model VR/080 (full span), DLCS vs. Alpha — BVWT 67, 12/8/70

STABILIZER CONTROL WITH CYCLIC
$i_w = 15°$, $i_p = 60°$
FULL SPAN SLATS
MIN $C_D = 150$

RUN	SYM	i	a	x
39	⬦	+5°	12.1	-4°
40	▷	0°		
41	▽	+10°		
42	+			

FOUR-PROP TILT WING
MODEL VR/080 (FULL SPAN)
DLCS VS. ALPHA

BVWT
67

12/ 8/70

Four-prop tilt wing model VROEBG (full span) — CMCG vs. Alpha. Stabilizer control with cyclic, $i_w = 36°$, $i_p = 60°$, full span slats, nom $C_{T_s} = .55$. Figure 85. Number D170-10089-1. Sheet 145.

RUN	SYM	△	q	Y
56	▽	+10°	8.2	+4°
57	▷	+5°		
58	+	----		
35&61	◁◁◁	+15°		

STABILIZER CONTROL WITH CYCLIC
iw = 30°   δ_F = 60°
FULL SPAN SLATS
NOM C_T_s = .55

10° Climb Angle

10° Descent Angle

Tail Off

FOUR-PROP TILT WING
MODEL VBC680 (FULL SPAN)
CLCS VS. CXCS

BVWT
67

12/10/70

SHEET 146

FOUR-PROP TILT WING
MODEL VRC680 (FULL SPAN)
CLCS VS. ALPHA

BVWT 67
12/10/70

NOT REPRODUCIBLE

FOUR-PROP TILT WING
MODEL VR06B0 (FULL SPAN)
CMC5 VS. ALPHA

Figure 88

NUMBER D170-10039-1
REV. LTR.

STABILIZER CONTROL WITH CYCLIC
$i_w = 30°$, $\delta_F = 60°$
FULL SPAN SLATS
NOM $C_{T_s} = .55$

SHEET 148

RUN	SYM	$\triangle$	q	γ
63	▽	+10°	8.2	-4°
64	◁	+5°		
67	+	---		
37&70	◁◁◁	+15°		

STABILIZER CONTROL WITH CYCLIC

$a_{W} = $? , $i_{T} = 50°$

FULL SPAN SLATS

NOM $C_T = .55$

FOUR-PROP TILT WING

MODEL VR0680 (FULL SPAN)

CLCS VS. CXCS

BVWT 67

12/10/70

SHEET 149

STABILIZER CONTROL WITH CYCLIC
i_w = 30°, δ_F = 60°
FULL SPAN SLATS
MOM C_T = .88

RUN	SYM	i_S	q	Y
63	▽	+10°	8.2	-4°
64	▷	+5°		
67	+	---		
37&70	◁	+15°		

CLUS

4.200

3.800

3.400

3.000

Tail Off

2.500

2.200

1.800

1.400

1.000

-24.000 -20.000 -16.000 -12.000 -8.000 -4.000 0.000 4.000 8.000

ALPHA - DEGREES

FOUR-PROP TILT WING	BVWT
MODEL VR6680 (FULL SPAN)	67
CLCS VS. ALPHA	12/10/70

SHEET 150

FOUR-PROP TILT WING
MODEL VRO688 (FULL SPAN)
CMCS VS. ALPHA

NUMBER D170-10039-1

STABILIZER CONTROL WITH CYCLIC
FULL SPAN SLATS

BVWT 67

NOT REPRODUCIBLE

STABILIZER CONTROL WITH CYCLIC
$i_w = 85°$, $δ_F = 60°$
FULL SPAN SLATS
NOM $C_{W_n} = 21$

RUN	SYM	ℓ	q	γ
168	⊙	+25	2.64	−6.0
169	⊡	+35	↓	↓

NUMBER D170-10039-1
REV. LTR.   Figure 92

10° Climb Angle →

10° Descent Angle

CLLS

DXCS

FOUR-PROP TILT WING	BVWT
MODEL VBCGB0 (FULL SPAN)	67
DLCS VS. DXCS	1/ 4/71

SHEET 152

STABILIZER CONTROL WITH CYCLIC

i_w=45°, δ_F=50°

FULL SPAN SLATS

NOM C_T = .81

RUN	SYM	i	q	γ
168	⊙	+25	2.64	-6.0
169	▣	+35	↓	↓

FOUR-PROP TILT WING

MODEL VROGSO (FULL SPAN)

CLLS VS. ALPHA

BVWT 67

1/4/71

6.5 LATERAL/DIRECTIONAL STABILITY IN TRANSITION

In this test, the lateral/directional stability characteristics
of the four prop, tilt wing model VRO68Q were investigated
through the transition flight regime both with and without the
vertical tail depicted in Figure 17. The positioning of the
tail on the fuselage, which resulted in a tail volume of .083,
is shown in Figure 15. This investigation was conducted with
the horizontal tail mounted on the fin in its high position
and with the stabilizer set at an angle representative for the
wing tilt position. A check was made with the horizontal tail
removed at low C_{T_S} values and the wing down to establish its
contribution to the vertical tail effectiveness. The possible
influence of cyclic pitch inputs on lateral/directional stabi-
lity was also evaluated.

The yaw runs of this investigation were performed at a constant
propeller RPM with selected combinations of wing tilt angle and
flap angle that corresponded to typical combinations required
through transition. All yaw data was acquired with zero fuse-
lage angle, an angle that is representative of the normal body
attitude in transitional flight. The off-setting of the model
from the vertical centerline of the tunnel test section as
shown in Figure 19, provided a yaw angular range of 8° (nose
right) to -24° (nose left). This value, the maximum nose left
yaw angle of -24°, was a limitation imposed by the allowable
bending of the braided steel air hose inside the yaw adapter.

All moments obtained during the lateral/directional runs were
transferred to the same wing down mid center of gravity loca-
tion and movement with wing tilt angle as used for the longitu-
dinal stability and control analysis. The body axis system was
used for the data presentations.

154

6.5.1 Empennage Off Lateral/Directional Stability

The empennage off lateral/directional stability characteristics as measured from the basic three component yaw data are summarized in Figure 94. This plot presents the slipstream yawing moment, rolling moment, and side force derivatives (with respect to the sideslip angle, β), as a function of the slipstream thrust coefficient. All slopes plotted in Figure 94 were read over a yaw angle range that did not exceed $\pm 8°$ and the C_{T_S} values were those prevailing at zero yaw angle.

It is apparent from an examination of the directional stability plot shown in Figure 94, that over the thrust coefficient range from zero C_{T_S} to 0.8 C_{T_S} which encompassed wing configurations from wing down and retracted flaps to 30° of wing tilt and 60° flaps, $C_{n_{S_\beta}}$ generally varies linearly as a function of $(1-C_{T_S})$ or freestream q. This variation indicates that the major contributor to the tail-off directional stability is the fuselage, a component that is acted upon primarily by freestream q. The reduction in yaw instability at 0.55 C_{T_S} with 30° of wing tilt is a special case in that wing stall was present.

A change from an unstable yaw condition to a stable condition that occurred at $0.82 C_{T_S}$ when the wing was tilted up from 30° to 45°, probably resulted to a large measure from a redistribution of the longitudinal pressures on the fuselage. The corresponding side force picture, wherein no change in side force was recorded when the wing was raised is consistent with this reasoning as is the observed change via tuft studies in the character of the flow aft of the wing at combinations of a high C_{T_S} value and wing tilt angles in the order of 45°. In this case, the increase in the destabilizing propeller side force and propeller yawing moment is largely offset by the respective shortening of the moment arm between the propeller hub and c.g., and the lessening of the propeller yawing moment acting about the aircraft yaw axis.

A positive dihedral effect (negative sign on C_{ℓ_β}) was exhibited by the model with the wing down, flaps up, and slats extended in the low C_{T_S} range evaluated. Deflecting the flaps to 40° increased the dihedral effect. It is surmised that the major portion of this increase was caused by a modification in the wing/body interference effects.

A large change in the $C_{\ell_{S_\beta}}$ from a negative value to a positive value, occurred at 0.2 C_{T_S} when the flap deflection was increased by 20° and the wing was tilted to 15°. A combination

155

of two factors could have produced this effect: the large
constant percent chord flaps have a swept forward hinge line
and the slipstream lift coefficient was increased by an incre-
ment of 2.15 as shown in Figure 95. The subsequent change in
dihedral effect with increasing C_{T_S} follows the decrease in
C_{L_S} illustrated in Figure 95.

Tilting the wing from 30° to 45° at 0.82 C_{T_S} increased the
dihedral effect. It is a reasonable assumption that this in-
crease primarily resulted from the change in spanwise lift
distributions on the inboard portion of the upwind and down-
wind wings along with the increase in propeller forces and
moments acting about the aircraft roll axis. Wing tilt
increases the propeller forces and moments, plus vertically
raises the vectors reacting about the aircraft rolling axis.

6.5.2 Empennage On Lateral/Directional Stability

Figure 96, presenting the empennage on lateral/directional
stability characteristics, was developed from the basic three
component yaw data in a similar manner to that used in
generating Figure 94. Additional yaw data, over that obtained
for the tail-off case, was acquired with a flap deflection of
40° in combination with the 30° of wing tilt. Likewise,
data was also acquired with 55° of wing tilt at 0.93 C_{T_S}.

The first item of interest in Figure 96 is the unstable direc-
tional characteristics measured with the wing down and
retracted flaps. This situation has been identified with a low
vertical tail effectiveness caused by the interaction of the
flow emanating from the wing/body juncture on the root section
of the vertical tail. Lowering the flaps to 40°, produced a
stable aircraft directionally by improving the vertical tail
effectiveness. Apparently, lowering the flaps increased the
vertical tail effectiveness by deflecting the adverse wing/
body juncture flow below the base of the fin. A redesign of
the wing/body intersection and vertical tail/fuselage junction
could probably rectify the deficiency.

At 0.2 C_{T_S}, the increase in directional stability accompanying
the 15° of wing tilt and additional 20° of flap deflection,
largely reflects a further increase in vertical tail effect-
iveness. The subsequent decrease in directional stability
with increasing thrust coefficient generally follows the
linear reduction in freestream q or dynamic pressure acting
on the vertical fin.

With a wing tilt angle of 30°, a reduction in flap deflection from 60° to 40° produced some increase in yaw stability and dihedral effect. These increases should have primarily resulted from the change in lift coefficient accompanying the flap angle change; however, the relatively large increase in dihedral effect does not appear to be consistent with the increment of lift coefficient illustrated in Figure 95.

The improvement in yaw stability at 0.93 C_{T_S} when the wing was tilted up to 55° from a 45° angle, is attributed to the redistribution of longitudinal pressures along the fuselage as mentioned previously in Section 6.5.1. At this thrust coefficient the vertical tail is essentially ineffective due to the low freestream q.

6.5.3 Vertical Tail Effectiveness

Vertical tail effectiveness through transition is presented in Figure 97 as an increment of yawing moment derivative in the slipstream coefficient format. The fin's contribution to the rolling moment and side force derivatives are also shown. These increments were directly obtained by subtracting the tail-off values noted in Figure 94 from the respective tail-on values plotted in Figure 96.

Figure 97 shows the large increase in tail effectiveness at low C_{T_S} values, discussed in Section 6.5.1, that occurred when the flaps were extended to the 40° position with the wing down. A further increase in fin effectiveness was produced when the wing was tilted up to the 15° setting. Section 6.5.1 pointed out that the wing/body juncture of the model was probably the prime factor in this problem.

With 15° of wing tilt, the decrease in tail effectiveness with increasing thrust coefficient is linear and generally follows a $(1-C_{T_S})$ variation. The effectiveness is thus a direct function of the freestream q. acting on the tail. Except for the special case of 0.55 C_{T_S} where wing stall plus center section stall and fin "blanketing" were present, the tail effectiveness with 30° of wing tilt angle showed no significant departure from the $(1-C_{T_S})$ variation. Some "blanketing" at the base of the fin occurred at 0.8 C_{T_S} when the wing was tilted up to 45°.

157

Figures 98 through 127 present the basic data from which the plots of lateral/directional stability and vertical tail effectiveness characteristics were derived. Data in slipstream notation is presented in sets of three plots, C_{n_S} vs β (yawing moment vs. sideslip angle), C_{ℓ_S} vs. (rolling moment), and C_{y_S} vs. β (side force), in the order of increasing wing tilt angle and flap angle. Each set of plots contains series of runs, empennage on and/or off, conducted at a constant propeller RPM (except the zero C_{T_S} runs) with a selected wing tilt/flap angle configuration. The three component data obtained for the wing down and retracted flap case at low values of slipstream thrust coefficient is plotted in Figures 98 through 100. As the wing was tilted up with the flaps extended, data was acquired over a range of thrust coefficients representative for the portion of the transition regime that was commensurate with the particular aircraft configuration. Figures 125 through 127 present data for the highest wing tilt angle evaluated (55°). This data was obtained at a thrust coefficient of 0.93 C_{T_S}.

In the following graphs, CYMCS, CYCS, and CRMCS, which represent body axis data, should be read as C_{n_S} (positive-nose right), C_{ℓ_S} (positive - right wing down), and C_{y_S} (positive-to the right), respectively. A positive sideslip angle is nose to the left.

158

BASIC LATERAL/DIRECTIONAL STABILITY
EMPENNAGE OFF

YAWING MOMENT

$i_w = 45°$, $\delta_F = 60°$

$i_w = 30°$, $\delta_F = 50°$

$(1 - C_{T_S})$ Variation

$C_{n_{s_\beta}}$

C_{T_S}

SYM	i_w	δ_F
●	0	0
◑	0	40°
□	15°	60°
△	30°	50°
◇	45°	60°

$i_w = 15°$, $\delta_F = 60°$

$i_w = 0°$, $\delta_F = 40°$

$i_w = 0°$, $\delta_F = 0°$

ROLLING MOMENT

$C_{\ell_{s_\beta}}$

C_{T_S}

SIDE FORCE

NOTES:
1. Model VRO68Q
2. Data from EVWT 067
3. Full Span Slats
4. Zero Fuselage Angle

$C_{Y_{s_\beta}}$

C_{T_S}
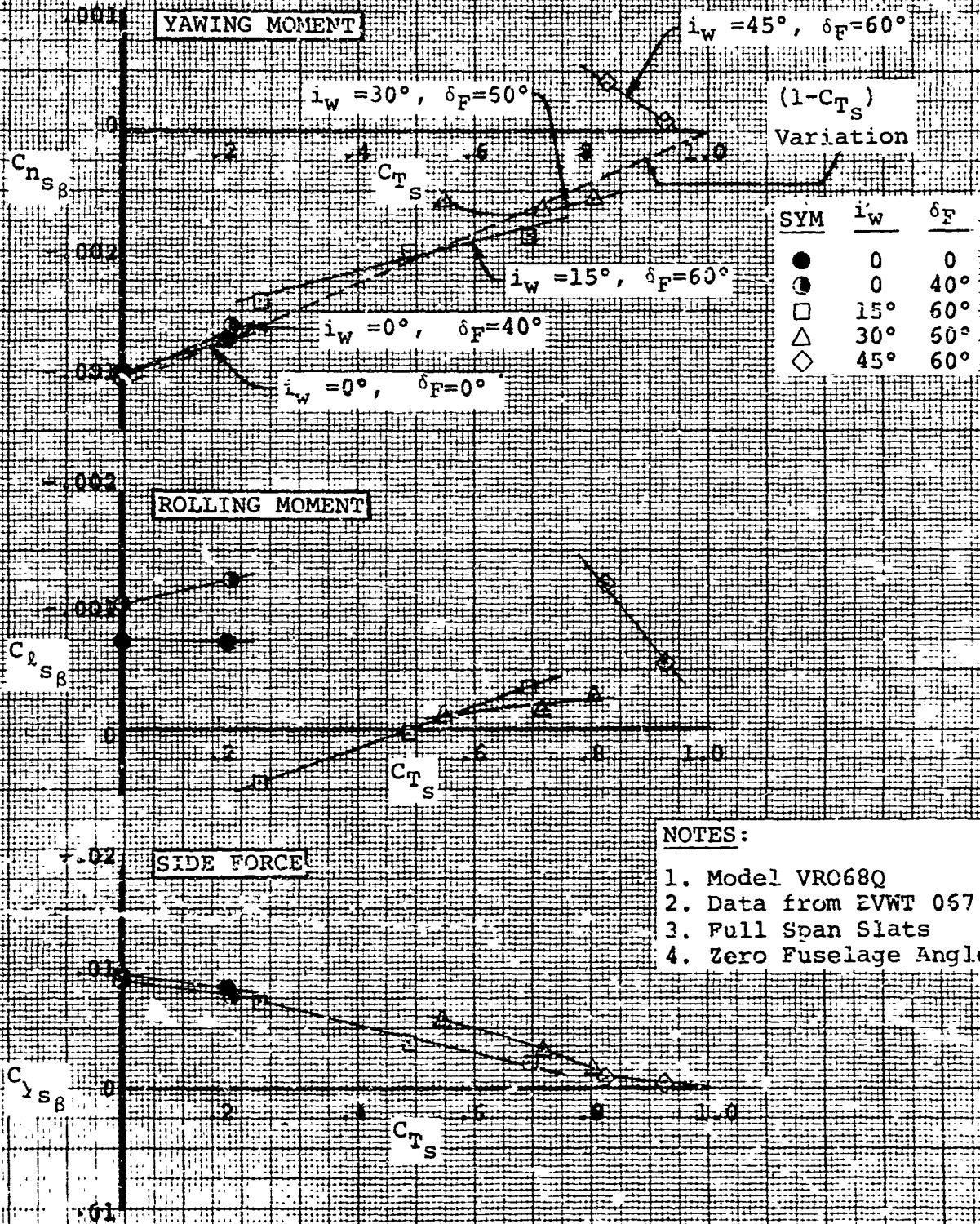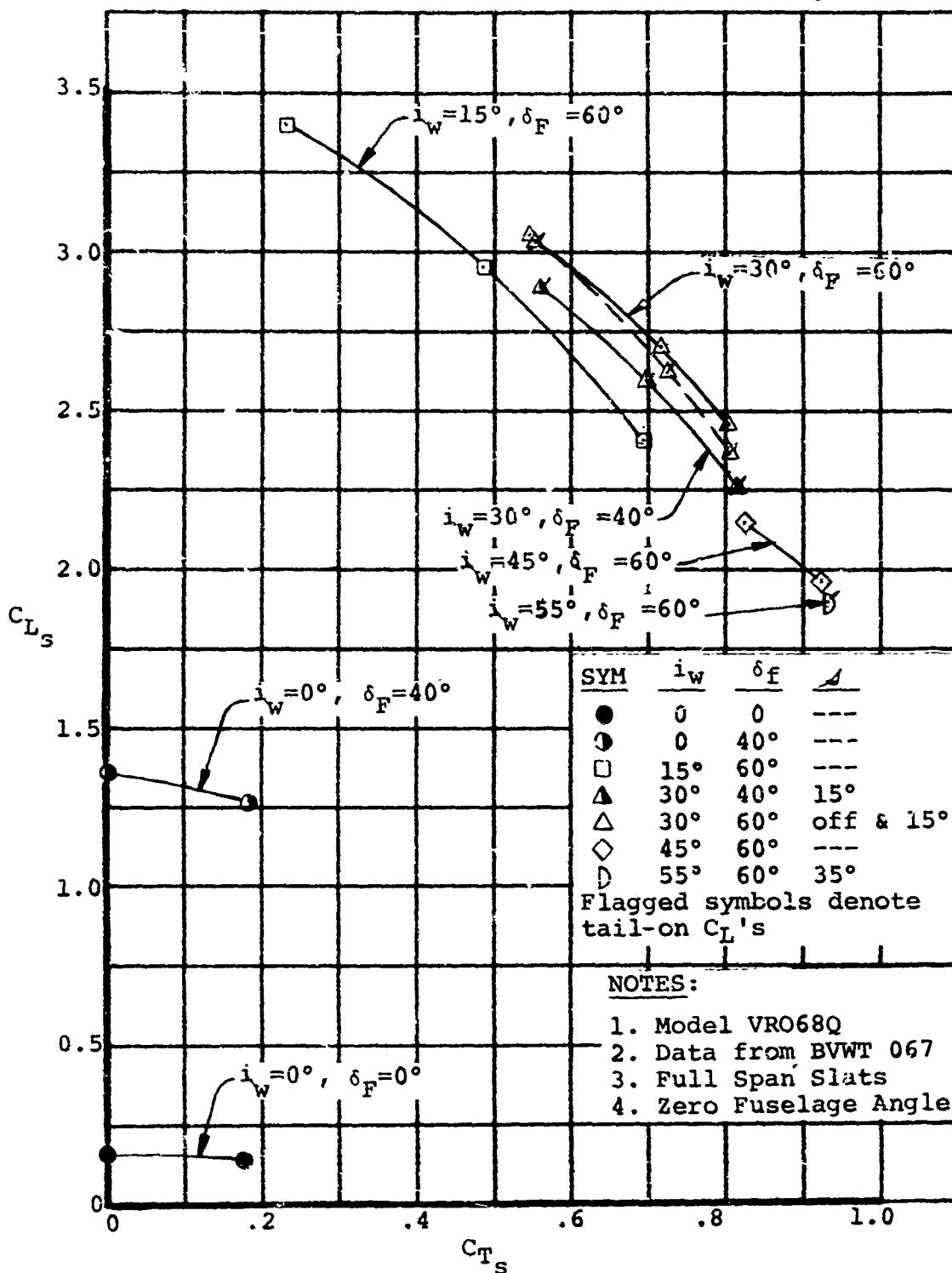
SLIPSTREAM C_L VARIATION
IN YAW TESTS

BASIC LATERAL/DIRECTIONAL STABILITY
EMPENNAGE ON

YAWING MOMENT
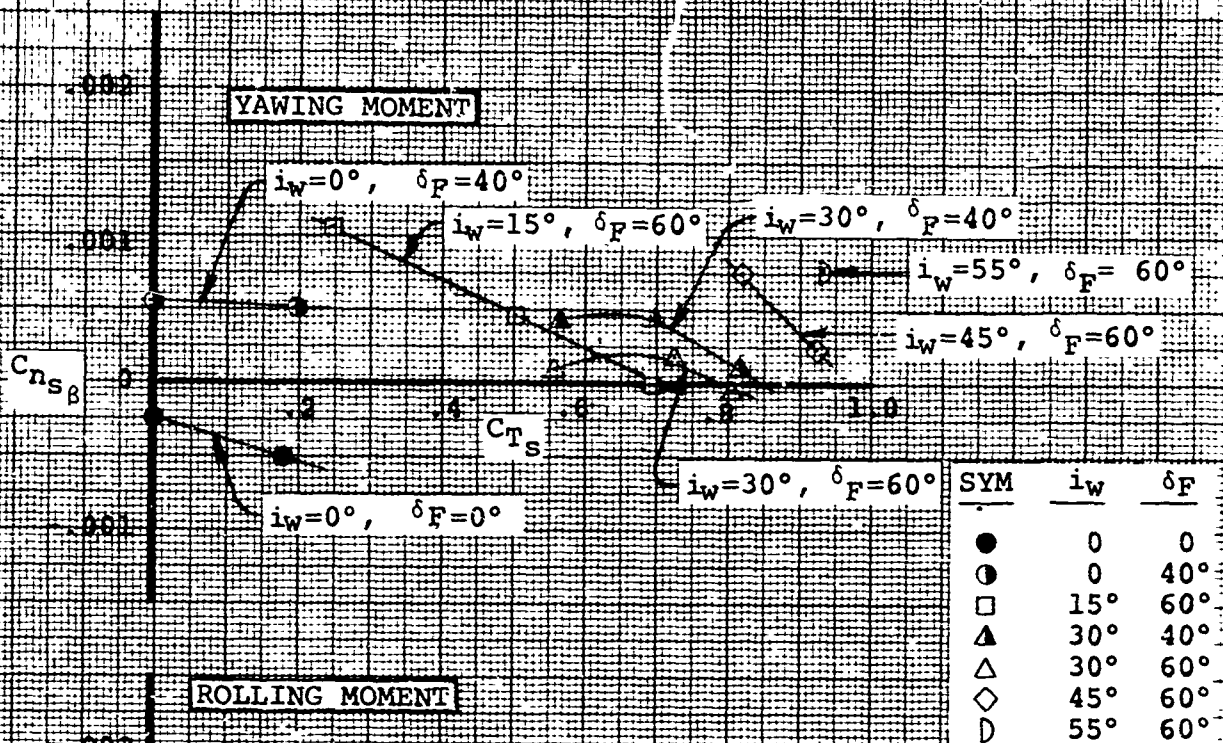
ROLLING MOMENT
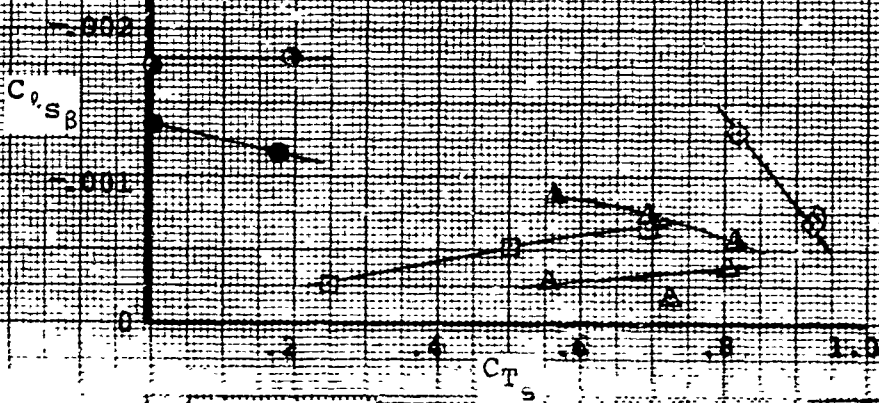
SIDE FORCE
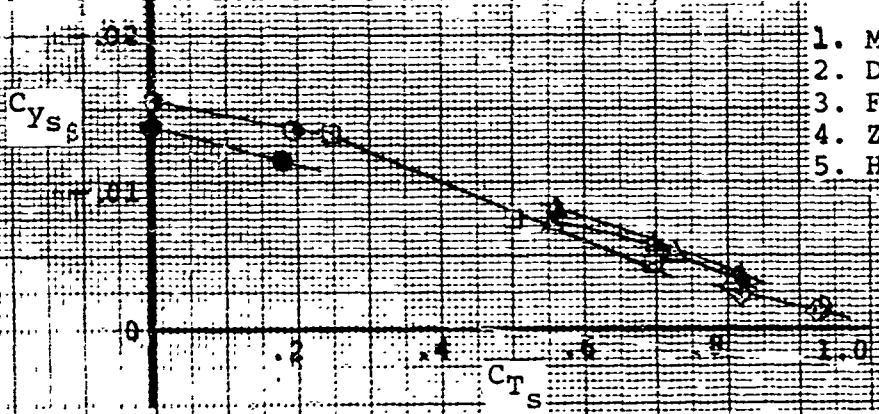
NOTES:

1. Model VRO68Q
2. Data from BVWT 067
3. Full Span Slats
4. Zero Fuselage Angle
5. Horizontal Tail On

SYM	i_w	δ_F
●	0	0
◑	0	40°
□	15°	60°
▲	30°	40°
△	30°	60°
◇	45°	60°
D	55°	60°

NOT REPRODUCIBLE

VERTICAL TAIL EFFECTIVENESS

YAWING MOMENT

SYM	i_w	δ_F
●	0	0
◑	0	40°
⊡	15°	60°
△	30°	60°
◇	45°	60°

$i_w=0°$, $\delta_F=40°$

$i_w=15°$, $\delta_F=60°$

$(1-C_{T_S})$ Variation

$i_w=30°$, $\delta_F=60°$

$i_w=45°$, $\delta_F=60°$

$\Delta C_{n_{S_\beta}}$

$i_w=0°$, $\delta_F=0°$

C_{T_S}

ROLLING MOMENT

$\Delta C_{\ell_{S_\beta}}$

C_{T_S}

SIDE FORCE

$\Delta C_{y_{S_\beta}}$

C_{T_S}
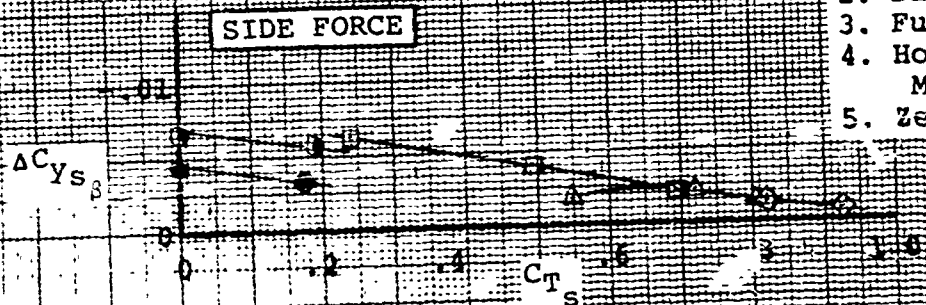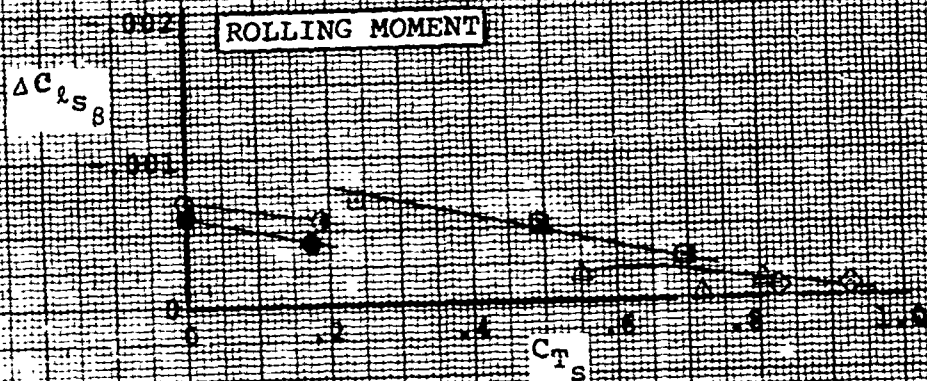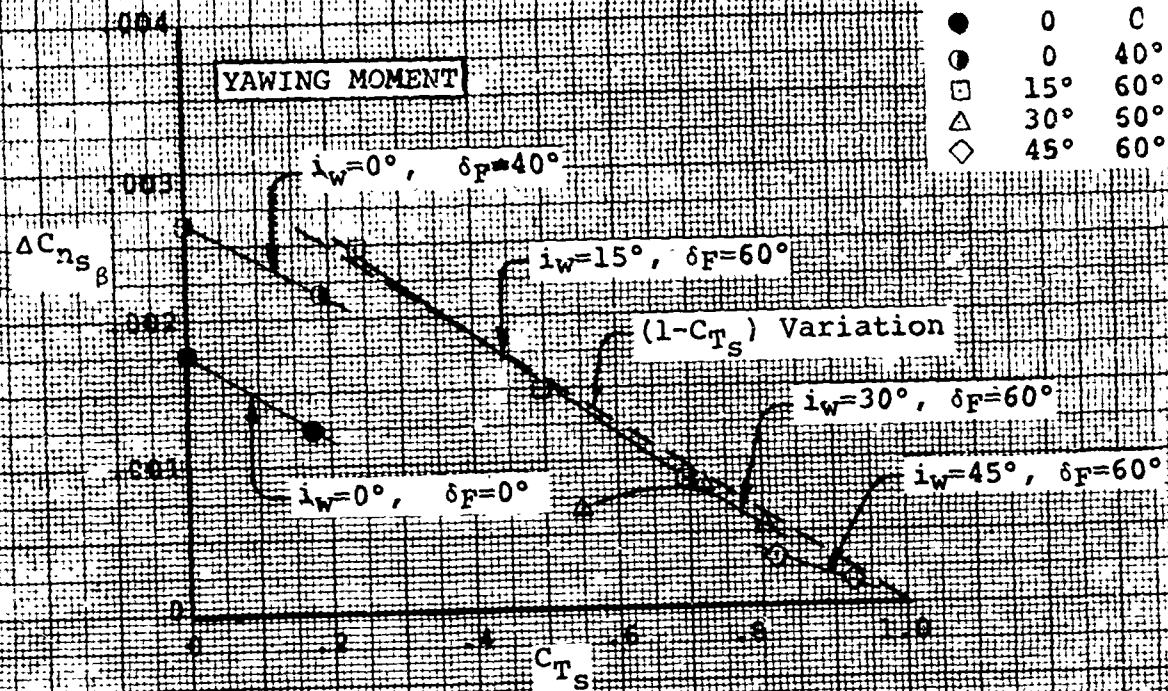
NOTES:
1. Model VRO68Q
2. Data from BVWT 067
3. Full Span Slats
4. Horizontal Tail
   Mounted on Fin
5. Zero Fuselage Angle

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MILLIMETER

LATERAL/DIRECTIONAL STABILITY
$i_w=0°$, $\delta_F=0°$
FULL SPAN SLATS

RUN	SYM	α	q	NOM CC_{T_S}
92	▷	0°	17.3	0
93	▽	0°	13.3	.20
94	◀	---	17.3	0
95	▲	---	13.3	.20

Empennage Off

Empennage On

CYMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR068B0 (FULL SPAN)	67
CYMCS VS. BETA	12/15/70

SHEET 163

RUN	SYM	δ	α	NOM C_{T_S}
92	▷	0°	17.3	0
93	▽	0°	13.3	.20
94	◀	---	17.3	0
95	▲	---	13.3	.20

Empennage Off

Empennage On

CYCS

0.200
0.100
0.000
-0.100
-0.200
-0.300
-0.400
-0.500
-0.600
-0.700

24.000 20.000 16.000 12.000 8.000 4.000 0.000 -4.000 -8.000

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR0689 (FULL SPAN)	67
CYCS VS. BETA	12/15/70

SHEET 164

LATERAL/DIRECTIONAL STABILITY
$i_w=0°$, $\delta_F=0°$
FULL SPAN SLATS

NUMBER D170-10039-1
REV. LTR. Figure 100

RUN	SYM	α	q	NOM C_{T_S}
92	▷	0°	17.3	0
93	▽	0°	13.3	.20
94	◀	----	17.3	0
95	▲	----	13.3	.20

FOUR-PROP TILT WING
MODEL VR065G (FULL SPAN)
CRMCS VS. BETA

BVWT
67
12/15/70

NOT REPRODUCIBLE

SHEET 165

FOUR-PROP TILT WING
MODEL VROGEO (FULL SPAN)
CYMCS VS. BETA

BVWT 67

1/ 4/71

LATERAL/DIRECTIONAL STABILITY
$i_w = 0°$, $\delta_F = 40°$
FULL SPAN SLATS

NUMBER D170-10039-1
REV. LTR. Figure 102

RUN	SYM	i_w	α	NOM C_{T_S}
85	▷	0°	17.3	0
86	▽	0°	13.3	.20
96	▶	--	17.6	0
97	▼	--	13.3	.20

NOT REPRODUCIBLE

FOUR-PROP TILT WING	BVWT
MODEL VROGEO (FULL SPAN)	67
CYCS VS. BETA	1/ 4.71

SHEET 167

SHEET 168

LATERAL/DIRECTIONAL STABILITY
$i_w=15°$, $\delta_F=60°$
FULL SPAN SLATS

RUN	SYM	∢	q	NOM $\overline{C}_{T_S}$
98	▶	---	13.3	.25
99	▼		8.2	.50
100	◀	▼	4.7	.70

BETA - ANGLE OF SIDESLIP

CYMCS

FOUR-PROP TILT WING		BVWT
MODEL VR068O (FULL SPAN)		67
CYMCS VS. BETA		12/15/70
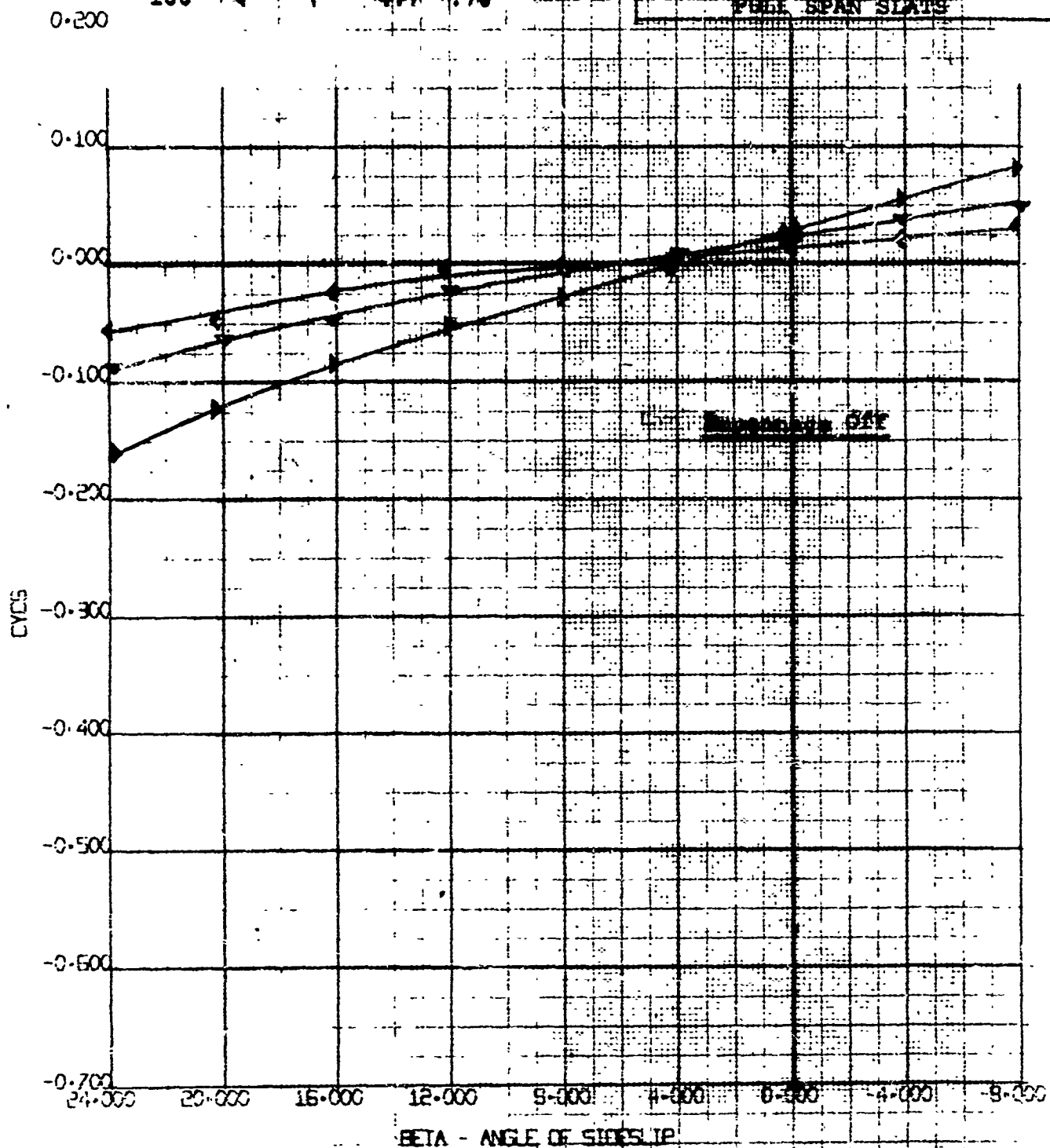
LATERAL/DIRECTIONAL STABILITY
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_s}
98		---	13.3	.25
99			9.2	.50
100			4.7	.70

NUMBER	D170-10039-1
REV. LTR.	Figure 103

FOUR-PROP. TILT WING
MODEL VROGBO (FULL SPAN)
CYCS VS. BETA

BVWT 67

12/15/70

SHEET 170
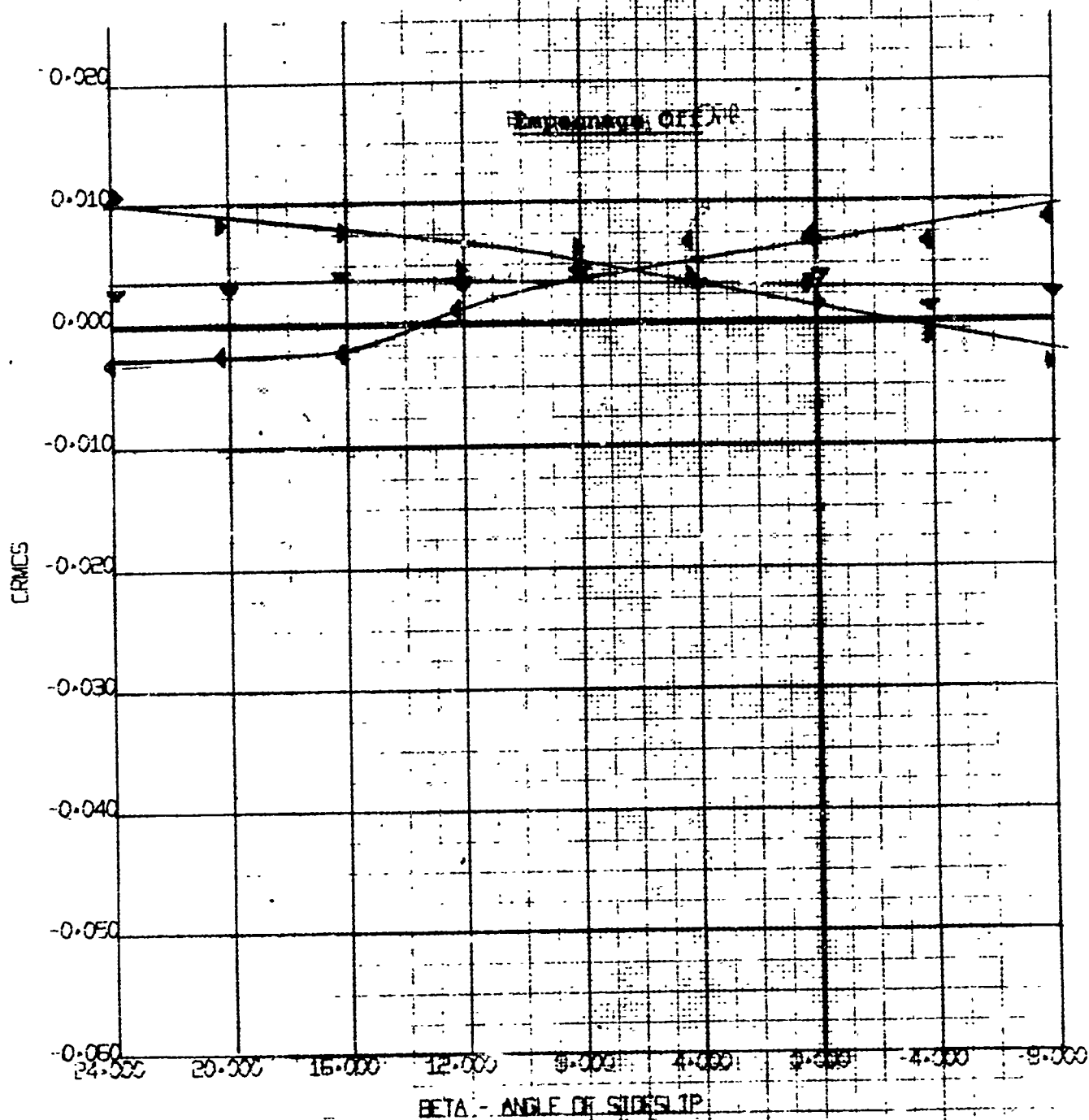
LATERAL/DIRECTIONAL STABILITY
$i_w = 15°$, $\delta_F = 60°$
FULL SPAN SLATS

RUN	SYM	α	q	NON C_{T_s}
76	△	+10°	13.3	.25
77	▷		8.2	.50
78	▽		4.7	.70

FOUR-PROP TILT WING
MODEL VROGGO (FULL SPAN)
CYMCS VS. BETA

	BVWT 67
	1/4/71

SHEET 172

Figure 108. LATERAL/DIRECTIONAL STABILITY $i_w = 15°$, $\delta_F = 60°$ FULL SPAN SLATS — CYCS VS. BETA

SHEET 173

FOUR-PROP TILT WING
MODEL VR686 (FULL SPAN)
CRMCS VS. BETA

SHEET 174

Figure 110

LATERAL/DIRECTIONAL STABILITY
$i_w = 30°$, $\delta_F = 40°$
FULL SPAN SLATS

RUN	SYM	α	q	NOM C_{T_S}
87		15°	9.2	.55
88			5.2	.70
89			3.3	.81

FOUR-PROP TILT WING
MODEL VRC680 (FULL SPAN)
CYMCS VS. BETA

BVWT 67
12/15/70

SHEET 175

FOUR-PROP TILT WING
MODEL VROBBC (FULL SPAN)
CYCS VS. BETA

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PAGE
IS
MISSING
IN
ORIGINAL
DOCUMENT

RUN	SYM		q	NOM C_{T_s}
101	▼		8.2	.55
102	▶		4.7	.70
103	◆		3.3	.81

LATERAL/DIRECTIONAL STABILITY
$i_w = 30°$, $\delta_f = 60°$
FULL SPAN SLATS

CYCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR0580 (FULL SPAN)	67
CYCS VS. BETA	12/14/70

SHEET 179
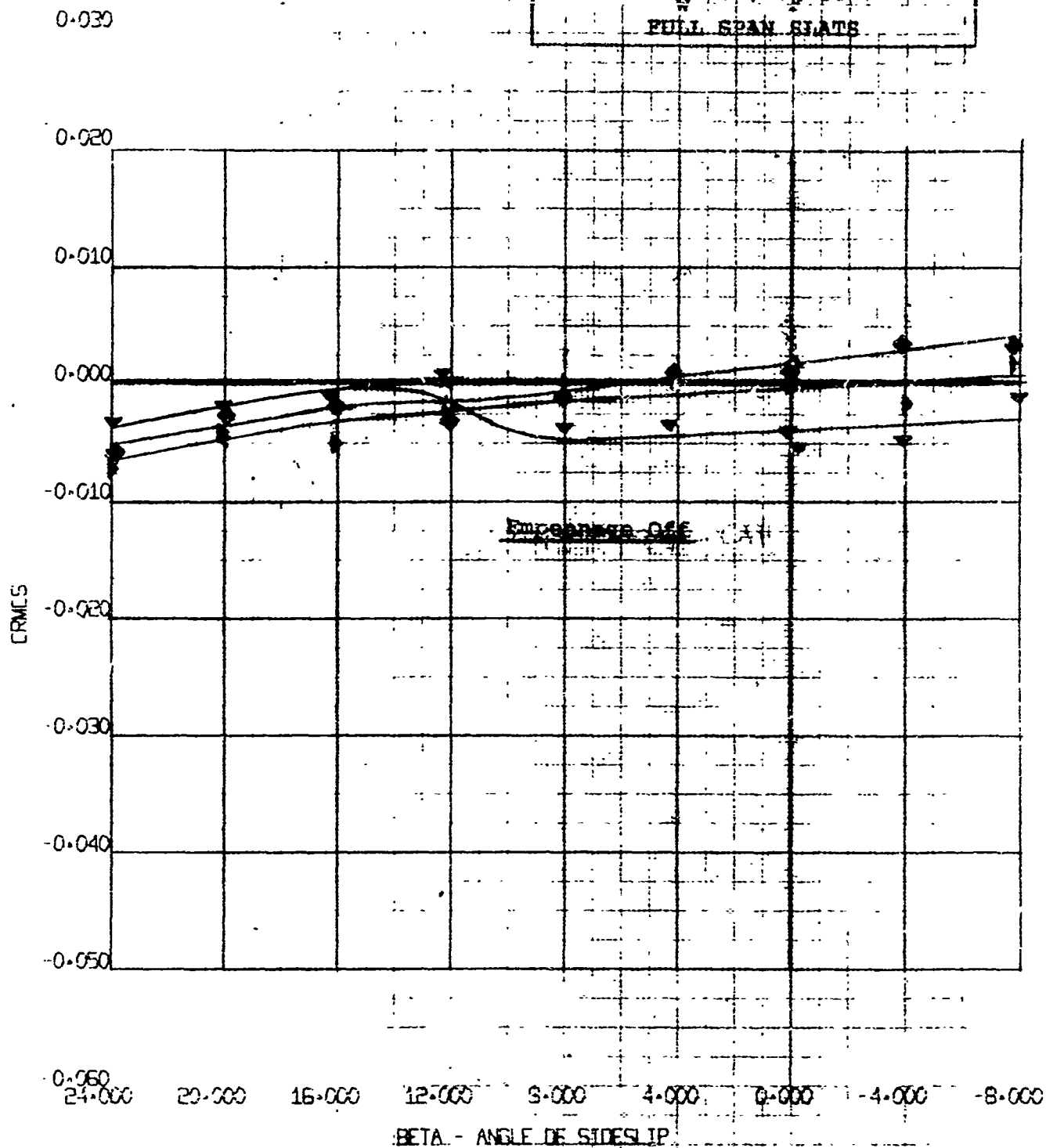
FOUR-PROP TILT WING
MODEL VROSBG (FULL SPAN)
CRMCS VS. BETA
BVWT 67
12/14/70

SHEET 180

LATERAL-DIRECTIONAL STABILITY

FULL SPAN SLATS

RUN	SYM		G	NOM CT
79	▽	15°	8.2	.55
80	▽		4.7	.70
81	◁		3.3	.81

CYMC5

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL WROGEO (FULL SPAN)	67
CYMC5 VS. BETA	1/4/71

SHEET 181

LATERAL/DIRECTIONAL STABILITY
$i_w = 30°$, $\delta_F = 60°$
FULL SPAN SLATS

RUN	SYM	α	q	NOM C_{T_S}
79	▷	15°	8.2	.55
80	▽		4.7	.70
81	◁		3.3	.81

Empennage On

BETA - ANGLE OF SIDESLIP

NOT REPRODUCIBLE

FOUR-PROP TILT WING	BVNT
MODEL VR0580 (FULL SPAN)	57
CYCS VS. BETA	1/4/71

SHEET 182

RUN	SYM	α	α	C_{T_s} NOM
79	▷	15°	8.2	.55
80	▽		4.7	.70
81	◁		3.3	.81

LATERAL/DIRECTIONAL STABILITY
$i_w=30°$, $\delta_F=60°$
FULL SPAN SLATS

Empennage On

CRMC5

BETA = ANGLE OF SIDESLIP

FOUR-PROP TILT WING
MODEL VRC580 (FULL SPAN)
CRMC5 VS. BETA

BVWT
67
1/471

SHEET 183

LATERAL/DIRECTIONAL STABILITY
$i_w=45°$, $\delta_f=60°$
FULL SPAN SLATS

RUN	SYM	α	q	NOM C_{T_s}
104	▼	25°	3.3	.81
105	◆	↓	1.3	.92

NUMBER D170-10039-1
REV. LTR. Figure 119

Empennage Off

BETA - ANGLE DE SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR0580 (FULL SPAN)	6'
CYMCS VS. BETA	12/15/70

SHEET 184

LATERAL/DIRECTIONAL STABILITY
$i_w = 45°$, $\delta_F = 60°$
FULL SPAN SLATS

RUN	SYM	i_w	α	NOM C_{T_s}
104	▼	25°	3.3	.81
105	◈	↓	1.3	.92

NUMBER D170-10039-1
REV. LTR. Figure 120

Empennage Off

BETA - ANGLE OF SIDESLIP

CYCS

FOUR-PROP TILT WING MODEL VROGBO (FULL SPAN) CYCS VS. BETA	BVWT 67
	12/15/70

LATERAL/DIRECTIONAL STABILITY
$i_w = 45°$, $\delta_p = 60°$
FULL SPAN SLATS

RUN	SYM	α	q	NOM C_{T_S}
104	▼	25°	3.3	.81
105	◆	↑	1.3	.92

Empennage Off

CRMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROB80(FULL SPAN)	67
CRMCS VS· BETA	12/15/70

SHEET 186

LATERAL/DIRECTIONAL STABILITY
i_w=45°, δ_F=60°
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_s}	γ
82	▷	25°	3.3	.81	
83	▽		1.3	.92	

NUMBER   D170-10039-1
REV. LTR.   Figure 122

Empennage On

CYMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	EVNT
MODEL VRC980 (FULL SPAN)	6"
CYMCS VS. BETA	12/14/70

NOT REPRODUCIBLE

SHEET 187

Figure 123. Lateral/Directional Stability — $i_w=45°$, $\delta_F=60°$ — Full Span Slats

NUMBER D170-10039-1
REV. LTR. Figure 123

RUN	SYM	δ	q	NOM C_{T_s}	y
62	▷	25°	0.3	.81	60
63	▽		1.3	.92	

Empennage On

CYCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
CYCS. VS. BETA

BVWT 67
12/14/70

SHEET 188

LATERAL/DIRECTIONAL STABILITY
i_W=45°, δ_F=60°
FULL SPAN SLATS

RUN	SYM	i_W	α	NOM C_{T_S}
62	▷	25°	3.3	.81
83	▽		1.3	.92

NUMBER D170-10039-1
REV. LTR. Figure 124

Empennage On

CRMCS

BETA - ANGLE OF SIDESLIP

NOT REPRODUCIBLE

FOUR-PROP TILT WING		BVWT
MODEL VR0680(FULL SPAN)		67
CRMCS VS. BETA		12/14/70

SHEET 189
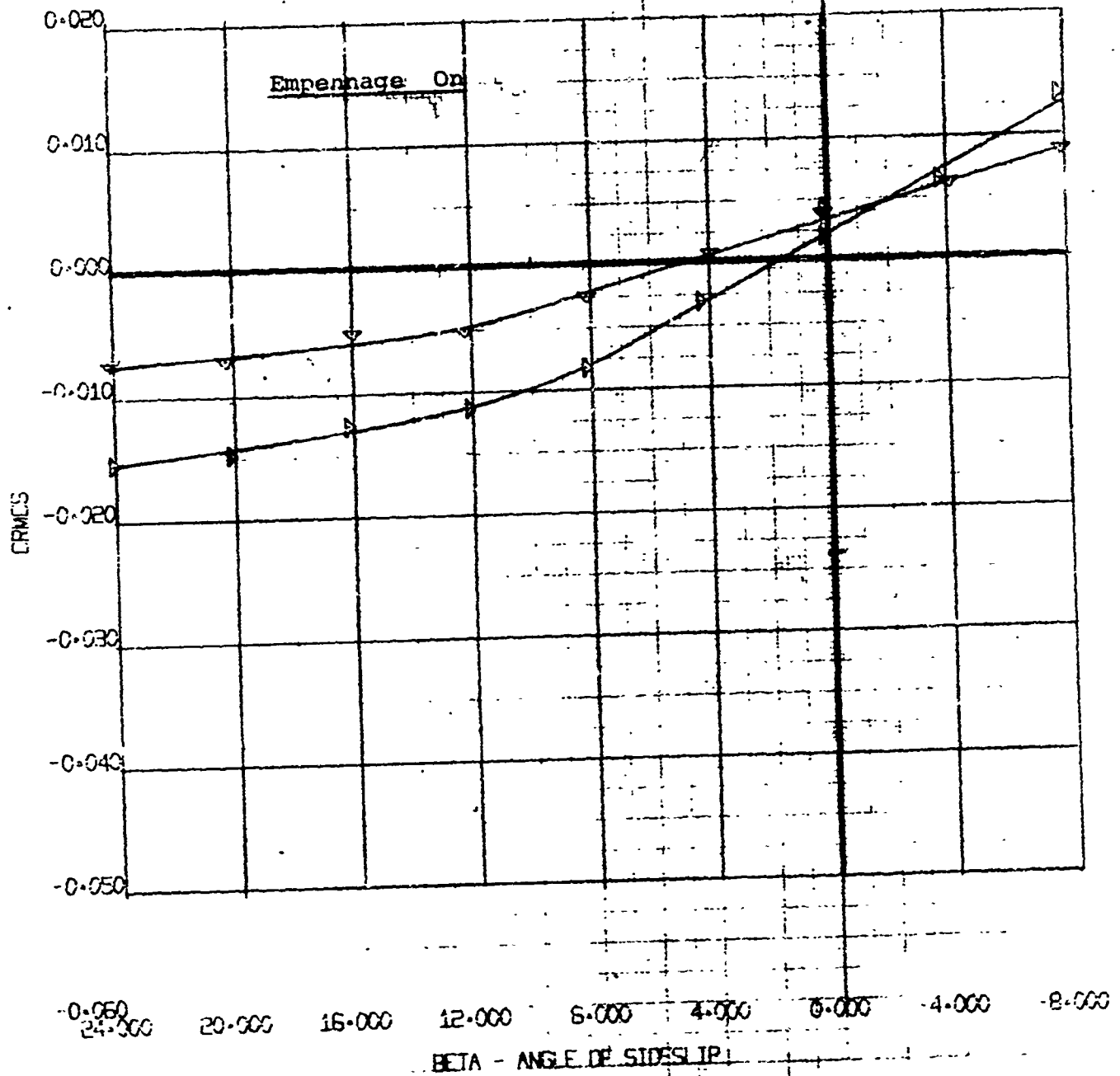
LATERAL/DIRECTIONAL STABILITY
$i_w=55°$, $\delta_F=60°$
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_S}	
84	▷	35°	1.16	.93	

NUMBER D170-10039-1
REV. LTR. Figure 125

Empennage On

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVW
MODEL VR0580 (FULL SPAN)	5'
CYMCS VS. BETA	12/11/70

SHEET 190

SHEET 191

RUN	SYM	$\dot{e}$	q	NOM C_{T_δ}	Y
84	$\triangleright$	35°	1.16	.93	

LATERAL/DIRECTIONAL STABILITY

$i_w = 55°$, $\delta_F = 60°$

FULL SPAN SLATS

0.030

0.020

Empennage On

0.010

0.000

-0.010

CRMCS

-0.020

-0.030

-0.040

-0.050

-0.060

| 24.000 | 20.000 | 16.000 | 12.000 | 8.000 | 4.000 | 0.000 | -4.000 | -8.000 |

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR06BC (FULL SPAN)	6?
CRMCS VS. BETA	12/14/7?

SHEET 192
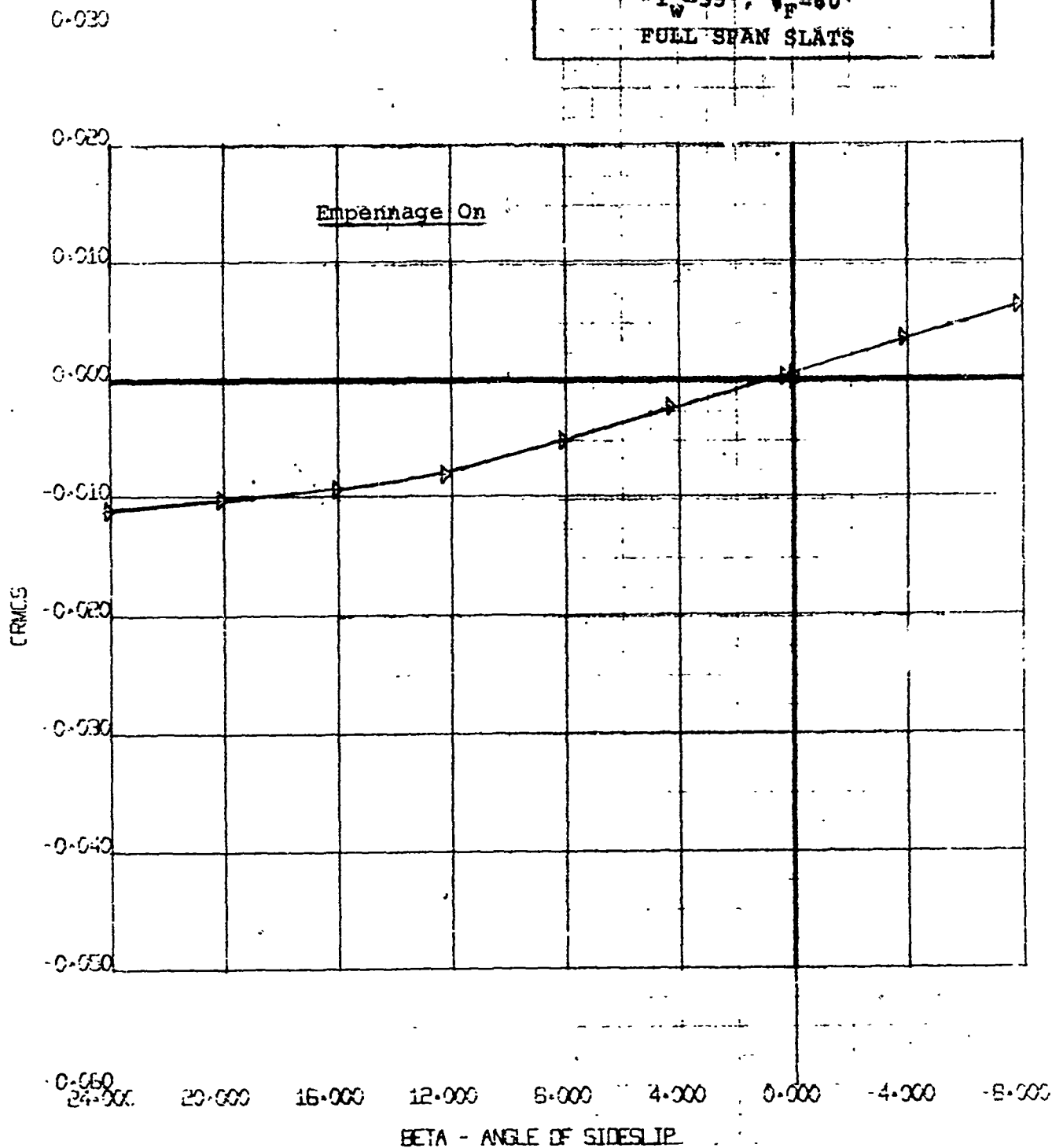
6.5.4 Effect of Horizontal Tail on Vertical Tail Effectiveness

The horizontal tail was mounted on top of the fin, as sketched
in Figure 15, during the empennage-on lateral/directional
stability testing. Stabilizer angles compatible with each
wing tilt angle were selected, i.e, the stabilizer angle was
manually set to a higher positive angle as the wing tilt angle
was increased. Since this tail configuration should have added
a substantial increment to the fin effectiveness, a check was
made to ascertain the magnitude.

Figure 128 presents the results of this investigation.
Compared in this figure are the lateral/directional stability
characteristics of the model for the wing down and 40° of flap
deflection case as measured with the horizontal on and off.
This data shows that mounting the stabilizer on top of the
vertical tail did provide some modest improvement in the yaw
stabilizing capability of the fin at the low thrust coeffic-
ients where the vertical tail effectiveness should be a
maximu. The magnitude of improvement is much less than the
25% anticipated, leading to the supposition that the fin's
effectiveness was low for the runs with the horizontal tail
off. This assumption is in agreement with Figure 97 and the
comments made in Sections 6.5.2 and 6.5.3 pertaining to
adverse flow conditions at the base of the fin emanating from
the wing/body juncture. It would be expected with this type
of flow problem which reduces the fin's effectiveness,
that the span loading on the fin would be significantly
altered over a major portion of the span. In this situation,
endplating the fin with the horizontal tail should not produce
the results that would normally occur.

The basic three component yaw data used in developing Figure
128 is presented in Figures 129 through 131. In these plots,
the solid lines/open symbols represent the base runs with the
horizontal tail installed and the dashed lines/solid symbols
represent the runs with the horizontal tail removed.

BASIC LATERAL/DIRECTIONAL STABILITY
HORIZONTAL TAIL ON & OFF
$i_w=0$, $\delta_F=40°$

YAWING MOMENT

SYM α

Horizontal Tail On

ROLLING MOMENT

SIDE FORCE

NOTES:

1. Model VR0680
2. Data from BVWT 067
3. Full Span Slats
4. Zero Fuselage Angle

SHEET 194

RUN	SYM	q	C_{T_s}	Δ
85	▷	17.3	0	0°
86	◁	13.3	.26	↓
90	▶	17.3	0	---
91	◀	13.3	.26	---

LATERAL/DIRECTIONAL STABILITY
$i_w=0°$, $\delta_F=40°$
FULL SPAN SLATS

-0.060

-0.000

-0.040

-0.020

-0.000

0.020

Horizontal Tail On and Off

0.040

0.060

0.060

0.100

| 24.000 | 20.000 | 16.000 | 12.000 | 8.000 | 4.000 | 0.000 | -4.000 | -8.000 |

BETA   ANGLE OF SIDESLIP

CYMCS

FOUR-PROP TILT WING	BVWT
MODEL VROGBO (FULL SPAN)	67
CYMCS VS. BETA	1 1 1

SHEET 195

LATERAL/DIRECTIONAL STABILITY
$i_w=0°$, $\delta_F=40°$
FULL SPAN SLATS

RUN	SYM	q	C_{T_S}	δ
85	▷	17.3	0	0°
86	◁	13.3	.26	↓
90	◆	17.3	0	---
91	◀	13.3	.26	---

Horizontal Tail On and Off

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	EVNT
MODEL VROGRC (FULL SPAN)	5
CYCS VS· BETA	1/ ⋅/71

LATERAL/DIRECTIONAL STABILITY
$i_w=0°$, $\delta_F=40°$
FULL SPAN SLATS

Horizontal Tail On and Off

RUN	SYM	α	C_{T_S}	Λ
85		17.3	0	0°
86		13.3	.20	
90		17.3	0	---
91		13.3	.20	---

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING
MODEL VROBBC (FULL SPAN)
CRLS VS. BETA

BVWT
67
1/4/71

SHEET 197
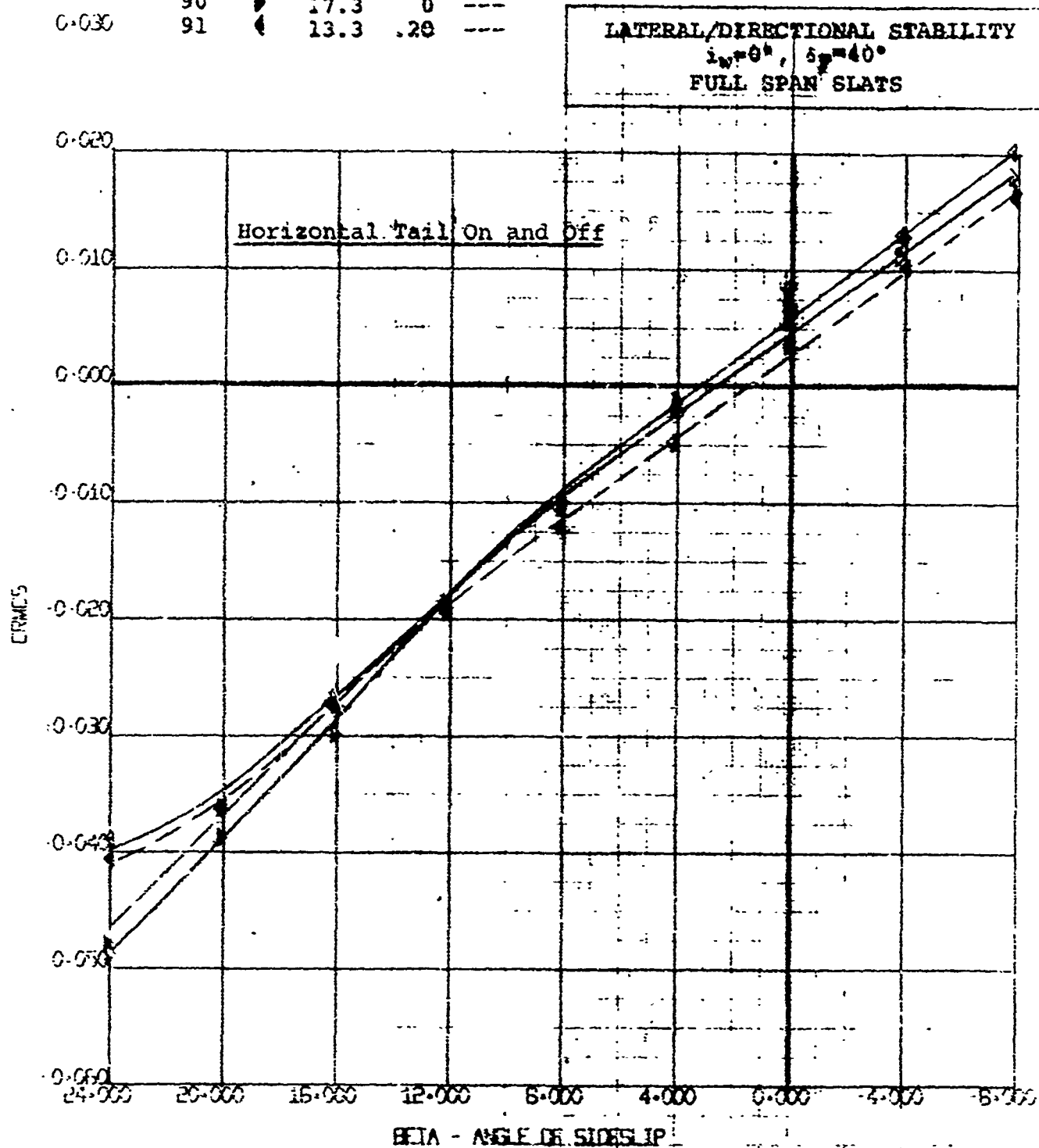
6.5.5 <u>Effect of Cyclic Pitch on Lateral/Directional Stability</u>

Following the investigation of basic lateral/directional stab-
ility characteristics, the effect of cyclic pitch on lateral/
directional stability was examined at selected points in tran-
sition with 15°, 30° and 45° of wing tilt plus 60° of flap
deflection. In Figure 132, the data acquired with the empennage
on and at zero fuselage angle is presented in the slipstream
derivative format. The zero cyclic fairings shown in Figure
132 are those previously shown in Figure 96.

Both positive and negative cyclic pitch angles are seen to have
only a minor effect on the directional stability ($C_{n_{s_\beta}}$) and
dihedral effect ($C_{\ell_{s_\beta}}$) at all conditions tested with positive
cyclic angles tending to decrease both the yaw stability and
dihedral effect and negative angles tending to increase them.
Thus, the cyclic pitch effects are in the same direction as
established for the longitudinal mode, indicating again the
influence of cyclic pitch on the propeller forces and moments.

Some data scatter is present in Figure 132, for example, the
yaw stability measured with 15° of wing tilt and zero cyclic
does not line up exactly with the comparable yaw stability data
acquired with both positive and negative cyclic pitch inputs.
This situation is understandable when it is realized that in
powered model testing, back-to-back runs for accuracy purposes
is not always feasible, because the length of a test series
can extend over a couple of days.

Figures 133 through 141 present the basic three component yaw
data from which the slopes shown in Figure 132 as data points,
were measured. The various sets of plots, arranged in order of
increasing wing tilt angle, use solid lines/open symbols to
represent positive cyclic angles and dashed lines/solid symbols
to represent negative cyclic angles. Slipstream thrust
coefficients selected for each wing tilt angle correspond to
typical points in transition; low thrust coefficients were
selected for evaluation of cyclic pitch effects with 15° of
wing tilt and a high thrust coefficient was selected with the
45° wing tilt angle.

One of the data accuracy problems associated with four-propeller powered model testing is the residual yawing moment encountered at zero yaw angle. As the wing is tilted up, this residual moment will be partially translated into a residual rolling moment . This data acquisition problem results from a combination of the inherent inaccuracy in individually setting the four propeller RPM's at each data point, the accuracy of each thrust calibration and the quality of thrust balancing achieved via the individual collective settings. An additional factor when air motors are used as in Model VR068Q is the jet thrust developed by each motor which can vary from motor to motor. The net result will be a displacement of the yawing moment curves and rolling moment curves about the zero yaw axis during the lateral/directional testing. This problem is particularly sensitive during the yaw testing due to the large moment arms involved. Zero shifts in the three component yaw data were minimized by incorporating subroutines into the data reduction program to subtract from the recorded yaw and roll moment, the contribution produced by the four individual jet thrusts and any unbalance in the individually measured propeller thrusts.

An examination of the three component yaw data plotted in Figures 98 through 127 and Figures 133 through 141 will reveal that this procedure was successful in most instances; however, in some data runs the tolerances must have buildup. Note that the scale used in plotting the rolling moment coefficient is twice that used for the yawing moment coefficient, which produced in some cases a larger apparent zero shift in the rolling moment data.

Figure 133. Lateral/Directional Stability, $i_w = 15°$, $\delta_F = 60°$, Full Span Slats. C_{YMC5} vs. Beta.

RUN	SYM	Δ	q	c_{T_s}	γ
106	◇	+10°	10.6	.25	+4°
107	△		6.3	.50	
117	◆		10.6	.25	+4°
118	▲		6.3	.50	

Empennage On

With Cyclic Pitch Control

FOUR-PROP TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CYMC5 VS. BETA	12/15/70

SHEET 201

LATERAL/DIRECTIONAL STABILITY
$i_w=15°$, $\delta_F=60°$
FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VZ066C (FULL SPAN)
CYCS VS· BETA

BETA - ANGLE OF SIDESLIP

With Cyclic Pitch Control

RUN	SYM	α	q	C_{T_S}	γ
106	◇	+10°	10.6	.25	+4°
107	▷		6.3	.50	
117	◆		10.6	.25	-4°
118	▲		6.3	.50	

Empennage On

0-030

LATERAL/DIRECTIONAL STABILITY
$i_W = 15°$, $\delta_F = 60°$
FULL SPAN SLATS

With Cyclic Pitch Control

CRMCS

BETA - ANGLE OF SIDESLIP

NOT REPRODUCIBLE

FOUR-PROP TILT WING	BVWT
MODEL VROBBO (FULL SPAN)	67
CRMCS VS. BETA	12/15/70

LATERAL/DIRECTIONAL STABILITY
$i_w = 30°$, $\delta_F = 60°$
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_s}	γ
108	△	+15°	6.5	.55	+4°
109	▽		4.0	.70	
119	◤		6.5	.55	-4°
120	◢		4.0	.70	

Empennage On
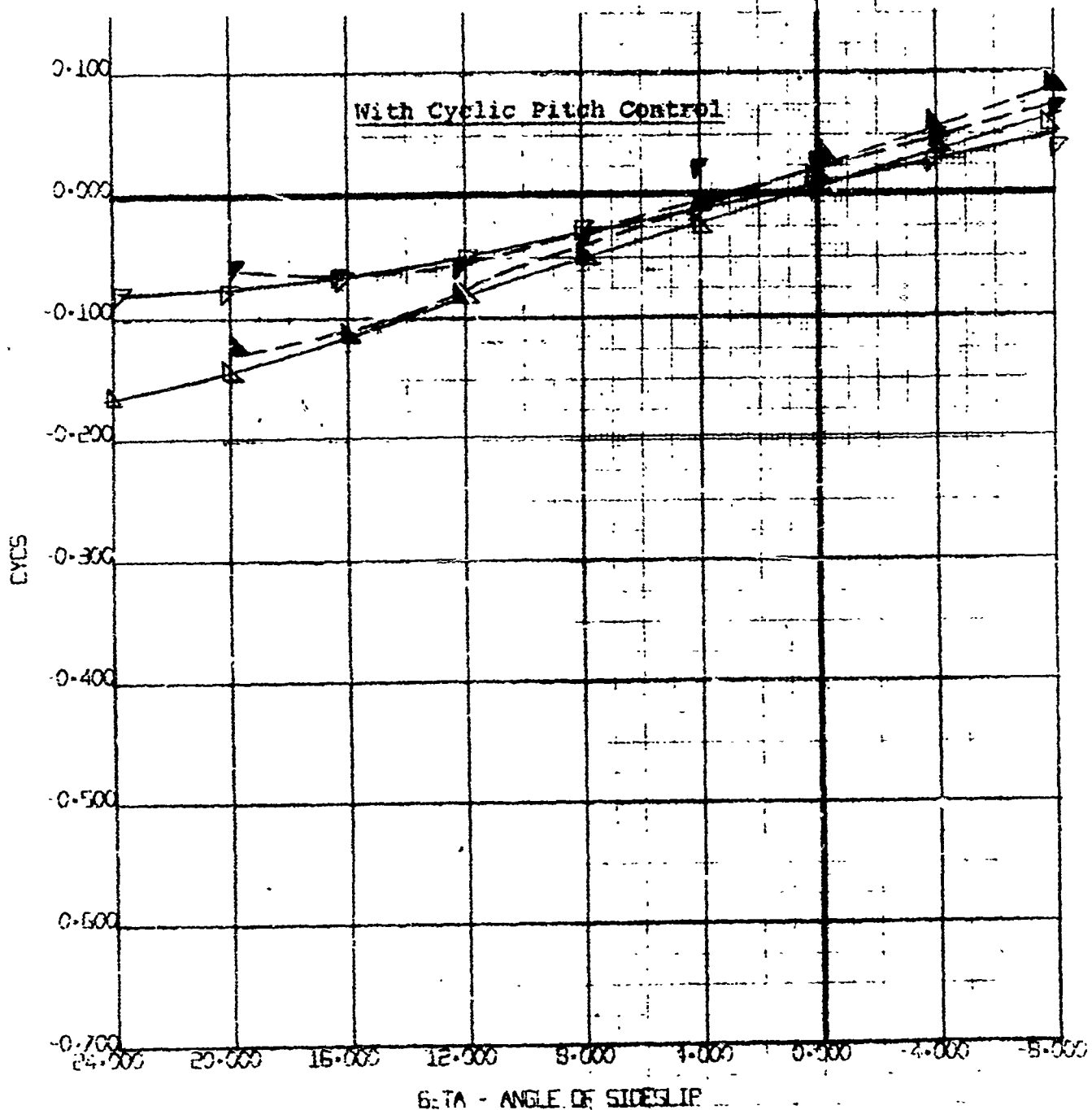
NUMBER D170-10039-1
REV. LTR. Figure 136

With Cyclic Pitch Control

BETA - ANGLE OF SIDESLIP

CYMC5

FOUR-PROP TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CYMC5. VS· BETA	12/15/70

SHEET 204

FOUR-PROP TILT WING
MODEL VROBBO (FULL SPAN)
CYCS VS. BETA

BVWT 67

12/15/70

FIGURE CONTENT:

NOM

RUN	SYM	Δ	α	$C_{T_{s}}$	γ
108	△	+15°	6.5	.55	+4°
109	▽		4.0	.70	
119	◀		6.5	.55	
120	▼		4.0	.70	

Empennage On

NUMBER D170-10039-1
REV. LTR. Figure 138

LATERAL/DIRECTIONAL STABILITY
$i_W = 30°$, $\delta_F = 50°$
FULL SPAN SLATS

With Cyclic Pitch Control

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR0681 (FULL SPAN)	67
CRMCS VS. BETA	12/15/70

SHEET 206

FOUR-PROP TILT WING
MODEL VRC580 (FULL SPAN)
CYMCS VS. BETA
BVWT 67
12/15/70

SHEET 207

FOUR-PROP TILT WING
MODEL VZ0680 (FULL SPAN)
CYCS VS. BETA

NOM
C_{T_s}

RUN	SYM	θ	α		i
110	⌄	+25°	2.7	.81	+4°
116	▼				-4°
172	△	+35°			+6°
167	▲	+25°			-6°

0·030

Empennage On

LATERAL/DIRECTIONAL STABILITY
$i_w=45°$, $\delta_F=60°$
FULL SPAN SLATS

With Cyclic Pitch Control

0·020

0·010

0·000

-0·010

-0·020

CRMCS

-0·030

-0·040

-0·050

-0·060

24·000   20·000   16·000   12·000   8·000   4·000   0·000   -4·000   -6·000

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROGBO (FULL SPAN)	67
CRMCS VS· BETA·	12/15/70

SHEET 209
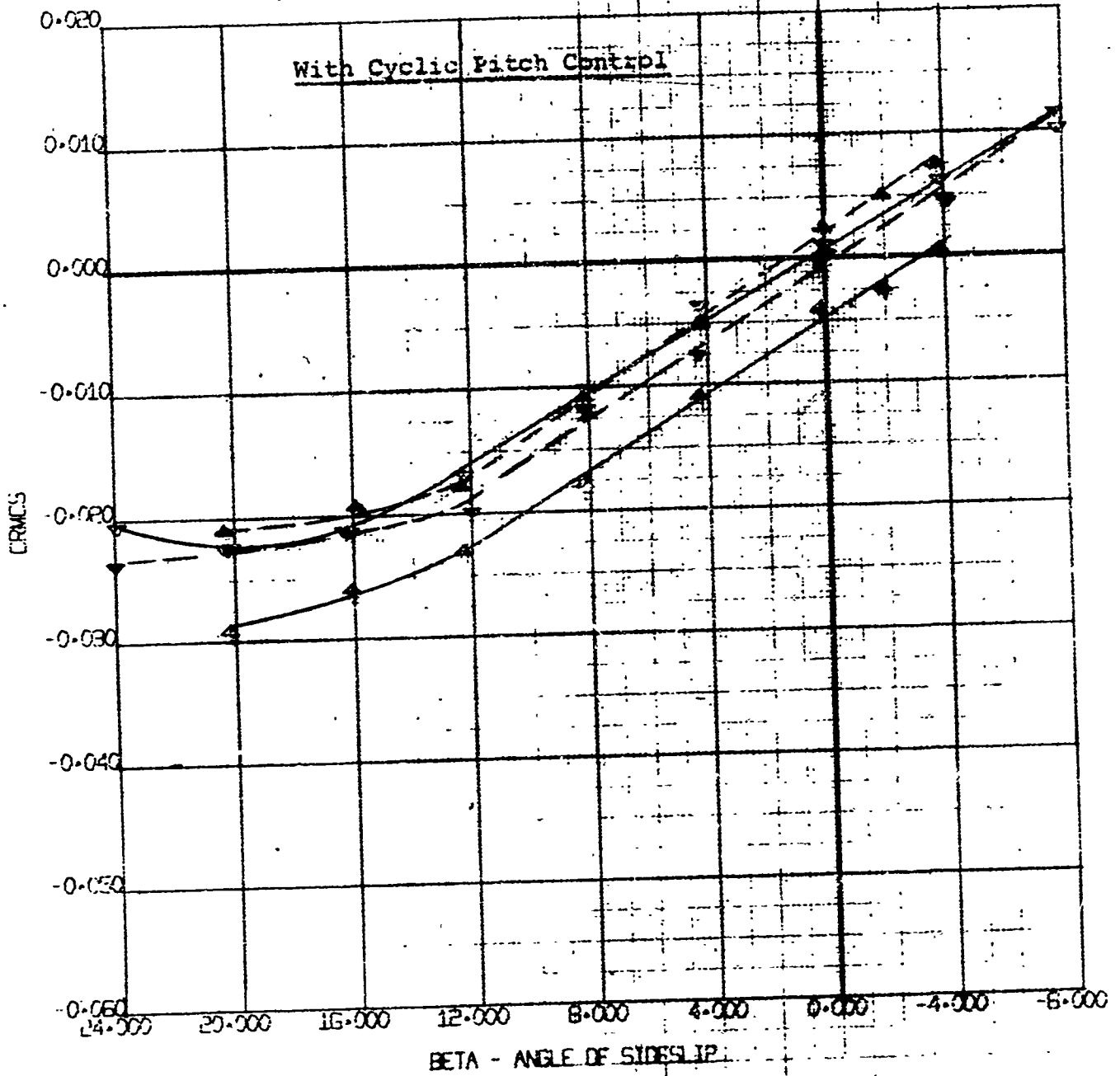
6.6   CROSS-COUPLING OF CYCLIC PITCH WITH ROLL/YAW SURFACE
      CONTROLS

Wing surface controls, used as a yaw control in hover with the
wing tilted about 90°, become a combined roll and yaw control
in transition where tilt angles can vary from about 45° at
0.81 C_{T_S} to 15° at 0.25 C_{T_S}. Roll and yaw control available
from a control configuration consisting of 20° differential
flaps and 40° spoilers was evaluated in fuselage pitch sweeps
at three selective wing tilt/thrust coefficient combinations
with a nominal 60° of flaps. The body axis data plotted in
Figure 142 in slipstream form shows the buildup in roll control
and phasing out of the yaw control as the wing is tilted down.
This is illustrated in the figure by the dashed line represent-
ing a constant propeller shaft angle. The variation of rolling
moment with fuselage angle is similar to the variation of
slipstream lift coefficient with fuselage angle.

Figure 143 which shows the changes associated with cross-
coupling cyclic pitch control and roll/yaw control includes
the data previously presented in Figure 142, but referenced
to the stability axis system instead of the body axis system.
Both positive and negative pitch inputs of 4° magnitude are
noted to have only a small percentage effect on the control
effectiveness.

Whereas Figure 143 illustrates the effect of cyclic pitch inputs
on the roll/yaw control in transition, Figure 144 presents the
opposing effect of the aircraft surface controls on the cyclic
pitch effectiveness. Cyclic pitch effectiveness with a coupled
roll/yaw control configuration of 20° of differential flaps and
40° of spoilers is shown in Figure 144 as an increment in slip-
stream pitching moment per degree of cyclic. Two typical
conditions in transition are compared with corresponding tail-
on data obtained during the cyclic pitch effectiveness runs
with a full 60° of flap deflection. The decrease in incremental
cyclic pitching moment capability with fuselage angle of attack
was mentioned earlier in the report as being largely a ramifi-
cation of the constant RPM method of testing, wherein slipstream
q is increased as the model was pitched.

Figure 144 indicates that at the 45° wing tilt/0.81 C_{T_S} condi-
tion, coupled roll/yaw control produced a change in cyclic
pitch effectiveness that varied from -2% to +10%. At the
30° wing tilt/0.55 C_{T_S} condition the change varied between
-3% and -7%. This total variation, though moderate in magni-

tude,does not appear to be consistent in its change in sign from a negative to a positive effect. A logical explanation would be that the overall effect of the application of roll/ yaw controls on cyclic pitching capability is small and that the results shown in Figure 144 merely reflect a normal amount of data scatter that occurs when tests extend over a period of several weeks and back-to-back run comparisons are not always possible. Such items as the accuracy of cyclic angular settings and changing ambient conditions in the test section are involved.

The basic pitching moment data used to develop the roll/yaw control coupled cyclic pitch effectiveness curves shown in Figure 144, is presented in Figures 145 and 146. This data is plotted as a slipstream pitching moment vs. fuselage angle of attack, with the moment being transferred about the previously described mid c.g. position. Figure 147 presents the corresponding variation of slipstream thrust coefficient for the constant RPM pitch runs.

The basic tail-on data without coupled roll/yaw control used to calculate the comparative cyclic pitch effectiveness curves shown in Figure 144 were presented earlier in the report in Figures 53 and 60.

ROLL/YAW CONTROL
$\delta_F = 40°/(LEFT)/60°(RIGHT)$
$\delta_s = 40°(LEFT)/0°(RIGHT)$

RUN	SYM	i_w	NOM C_{T_s}
125	◐	15°	.25
126	◑	30°	.55
127	○	45°	.81

$i_w = 45°$

$\alpha_p = +25°$

$i_w = 30°$

C_{n_s}

YAW CONTROL

$i_w = 15°$

$\alpha_F \sim$ DEGREES

BODY AXIS SYSTEM

NOTES:
1. Model VRO68Q
2. Data from BVWT 067
3. Full Span Slats
4. Stabilizer angle: +20°

$\alpha_F \sim$ DEGREES

ROLL CONTROL

$i_w = 45°$

$\alpha_p = +25°$

C_{ℓ_s}

$i_w = 30°$

$i_w = 15°$

EFFECT OF CYCLIC PITCH
ON ROLL/YAW CONTROL
δ_F=40° (LEFT)/60° (RIGHT)
δ_S=40° (LEFT)/0° (RIGHT)
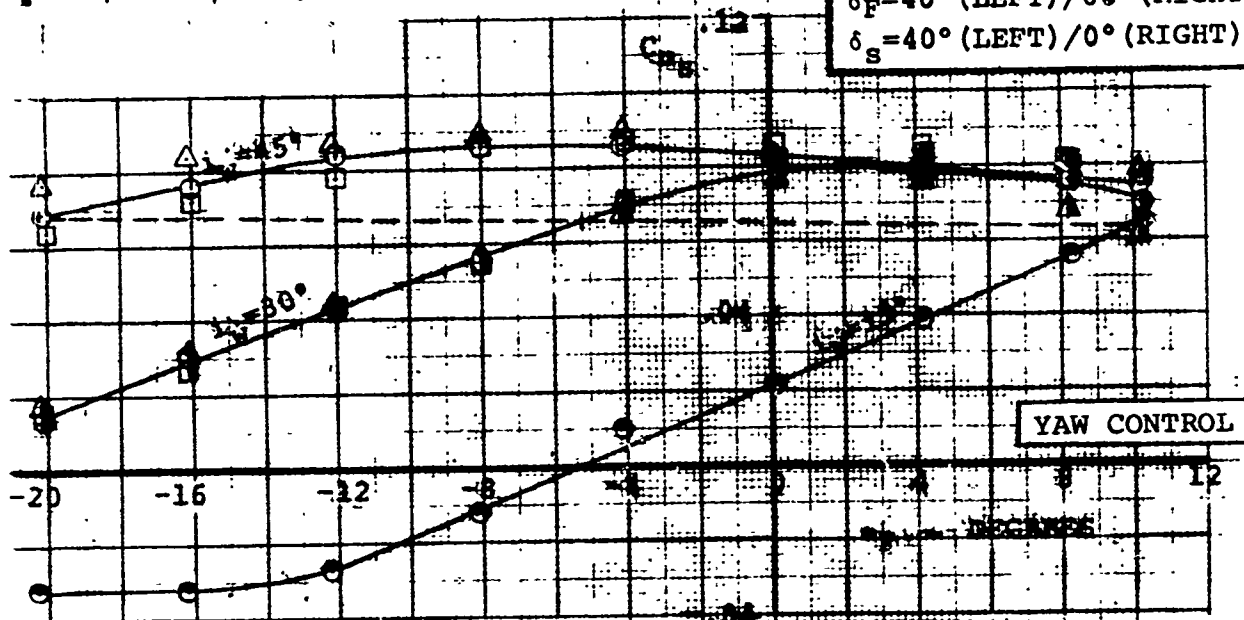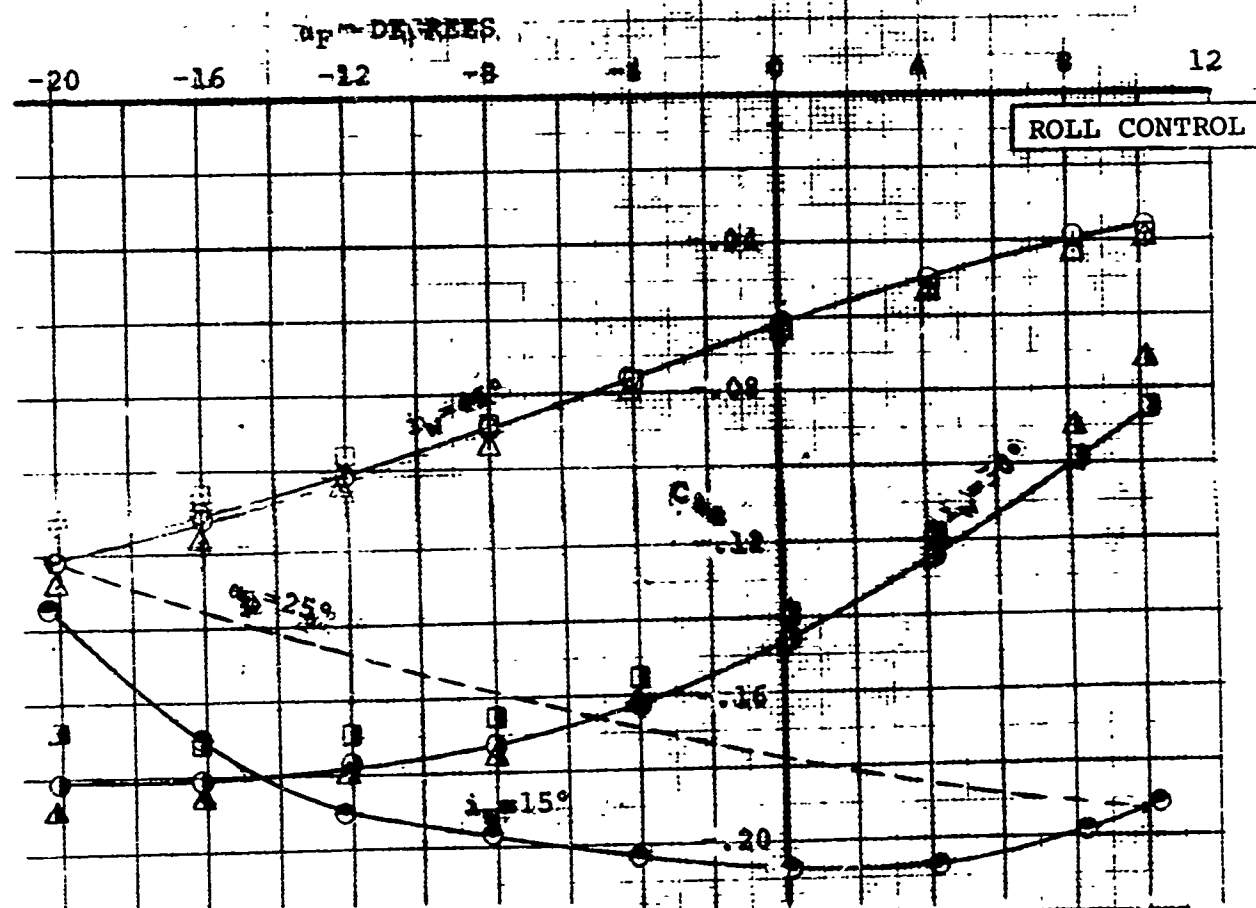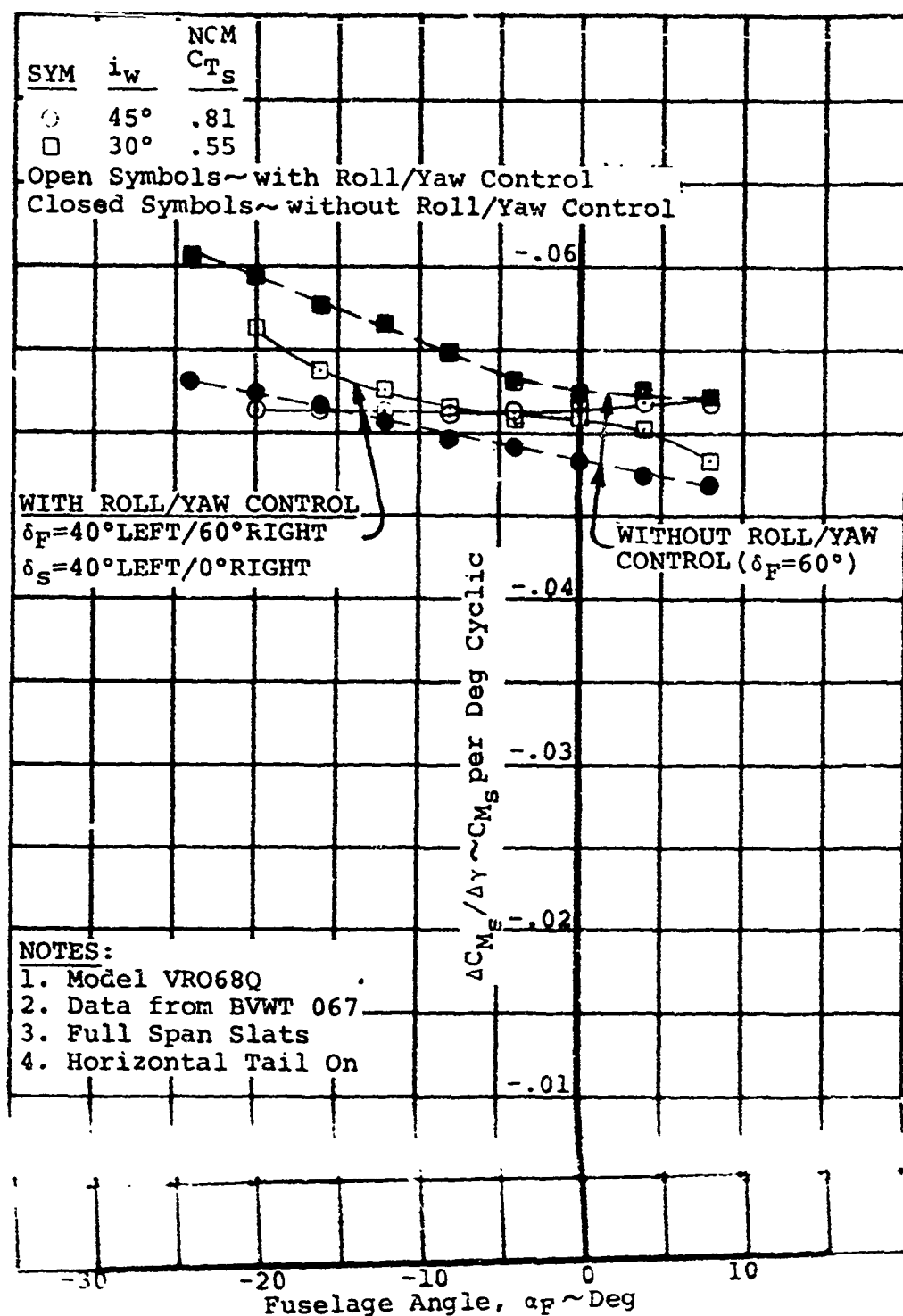
EFFECT OF ROLL/YAW CONTROL
ON CYCLIC PITCH EFFECTIVENESS

214

CYCLIC PITCH WITH ROLL/YAW CONTROL
$i_w=30°$, $\delta_F=40°$, LEFT/60° RIGHT
$\delta_H=40°$ LEFT/0° RIGHT
FULL SPAN SLATS
NOM $C_{T_S}=.55$

RUN	SYM	α'	q	γ
123	△	20°	6.5	+4°
126	○	↓	↓	0
121	□	↓	↓	−4°

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
CMCS VS. ALPHA

BVWT 67

1/4/71

RUN	SYM	i_w	q	γ
124	△	20°	2.7	+4°
127	O	↓	↓	0
122	□	↓	↓	-4°

CYCLIC PITCH WITH ROLL/YAW CONTROL
i_w=45°, δ_B=40°, LEFT/60° RIGHT
δ_S=40° LEFT/0° RIGHT
FULL SPAN SLATS
NOM C_{μ_S} =.81

CMC5

ALPHA - DEGREES

FOUR-PROP TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CMC5 VS. ALPHA	1/ 4/71

SHEET 216

VARIATION OF C_{T_S} WITH FUSELAGE ANGLE
FOR CROSS COUPLING RUNS
4500 RPM

6.7 LONGITUDINAL CHARACTERISTICS IN GROUND EFFECT

An investigation of the basic longitudinal characteristics in ground effect and an evaluation of cyclic pitch effects on the longitudinal characteristics were performed with the full-span model utilizing the Boeing-Vertol moving ground plane. This investigation was conducted with the aircraft level and a scaled ground height of 1.25 h/D representing the aircraft with the wheels 3 feet off the ground. Slats were extended and the flaps were deflected both 40° and 60° for the longitudinal runs. The 40° flap setting corresponds to a compromise between a lower deflection for takeoff with a wing tilt angle in the order of 20° and a higher deflection for landing approach.

The desire to obtain in-ground effect information at a low wheel height precluded performing the longitudinal mode runs with the standard fuselage angle pitch sweeps and constant wing tilt angles as a result of the clearance problem between the long straight sting and the moving ground plane. Wing tilt sweeps. with a level fuselage and tail removed, that were made in place of the fuselage sweeps do not allow a direct data comparison to show the influence of ground effect on the aircraft longitudinal stability. The level fuselage corresponds to a normal fuselage attitude for a tilt wing aircraft operating in close proximity to the ground.

6.7.1 Influence of Ground on Out-of-Ground Effect
       Longitudinal Characteristics

Longitudinal characteristics were investigated in ground effect with both 40° and 60° of flap deflection over a range of thrust coefficients from 0.25 to 0.81 C_{T_S} (nominal values) using a propeller operating speed of 4500 RPM. Figures 148 and 149 provide a direct comparison of the lift and force polars, respectively, for the two flap settings.

Figure 148 compares the slipstream lift coefficient developed as a function of the slipstream thrust coefficient at wing incidence angles of 0° and 10°. A maximum C_{L_S} comparison is also shown. This figure, which illustrates the typical decrease in slipstream lift coefficient with an increase in slipstream thrust coefficient, shows that at zero wing incidence an additional 20° of flap deflection beyond the 40° setting produces a substantial increment in lift. The 0.82 C_{L_S} increment at

218

0.18 C_{T_S} agrees favorably with the 0.81 C_{L_S} increment
previously obtained for the same configuration change during
out-of-ground effect testing at zero C_{T_S}. (See Figure 68 of
Reference 1.)

The increment in lift, from the change in flap angle, decreases
with increasing wing incidence angle such that at maximum lift,
the increment is reduced to 0.08 C_{L_S}, a value that is one-
third the value obtained out-of-ground effect at zero C_{T_S}.
As usual with flaps, the angle of attack at maximum lift was
less with 60° of flap than with 40° of flap. This situation
can sometimes completely cancel any incremental lift improve-
ment resulting from an increase in flap angle even though
full span slats are employed.

An additional point of interest in Figure 148 is the sharp
drop in maximum lift that occurs at 0.8 C_{T_S} with the 60° flap
setting. This condition is identified later in the report
with ground recirculation.

Figure 149 provides a direct comparison of the force polars
obtained in-ground effect with 40° and 60° of flaps. Except
for the decrease in lift occurring as a result of ground
recirculation at 0.81 C_{T_S} and 30° of wing tilt angle with the
60° flap angle, the force polars exhibit the normal trend
usually associated with a flap change to a larger angle:
higher drag or less propulsive force at a low wing angle of
attack and more lift at the high wing angles. The propulsive
force change at low wing incidence angles is produced by a
combination of increased flow turning and higher drag with
the larger flap angle.

Figures 150 and 151 were developed to illustrate the
influence of ground proximity on the lift curves and force
polars with 60° of flap deflection. In these figures, the
in-ground effect data is plotted as acquired during the
Phase II testing. The out-of-ground data as presented, is the
result of cross-plotting the data previously presented in
Figures 62 and 63 of Reference 1, but in another coefficient
form. Reference 1 data in the slipstream format, that was
acquired during similar wing tilt sweeps with a comparable
model configuration of flaps/slats, cyclic hubs and 4750 RPM,
was cross-plotted to provide data for a direct comparison at
the same slipstream thrust coefficients.

219

The lift curve comparison presented in Figure 150 shows an increase in lift at the three foot wheel height with the wing down. This lift increase due to ground effect was a maximum at the lowest thrust coefficient evaluated. Maximum lift in-ground effect was less than for the out-of-ground case; a condition primarily induced by the lower wing incidence angles involved. An examination of the curves will reveal that except for the ground recirculation incident at 0.81 C_{T_S}, the loss in maximum lift due to ground effect is essentially a constant increment over the C_{T_S} range tested.

Again, except for the 0.81 C_{T_S} ground recirculation condition, ground proximity has an effect on the O.G.E. force polars which can be considered typical. At the lower wing incidence angles, the effect of the ground plane is to reduce the drag by an increment of drag that varies inversely with the slipstream thrust coefficient.

Basic three component in-ground effect data with the two different flap deflections is presented in Figures 152 through 154 and Figures 155 through 157 for the 40° and 60° flap settings, respectively. The force polars, lift curves, and pitching moment curves are shown in the slipstream coefficient system used throughout the report. As with the other wing tilt runs, the pitching moments have been referred to the wing pivot located 42.6% MAC aft of the wing leading edge and 11.7% MAC below the wing chord plane. Figure 158 illustrates the variation in slipstream thrust coefficient with wing tilt angle that occurred during the constant RPM runs.

NOTES:
1. Model VRO68Q
2. Data from BVWT 067
3. Full Span Slats
4. Horizontal Tail Off
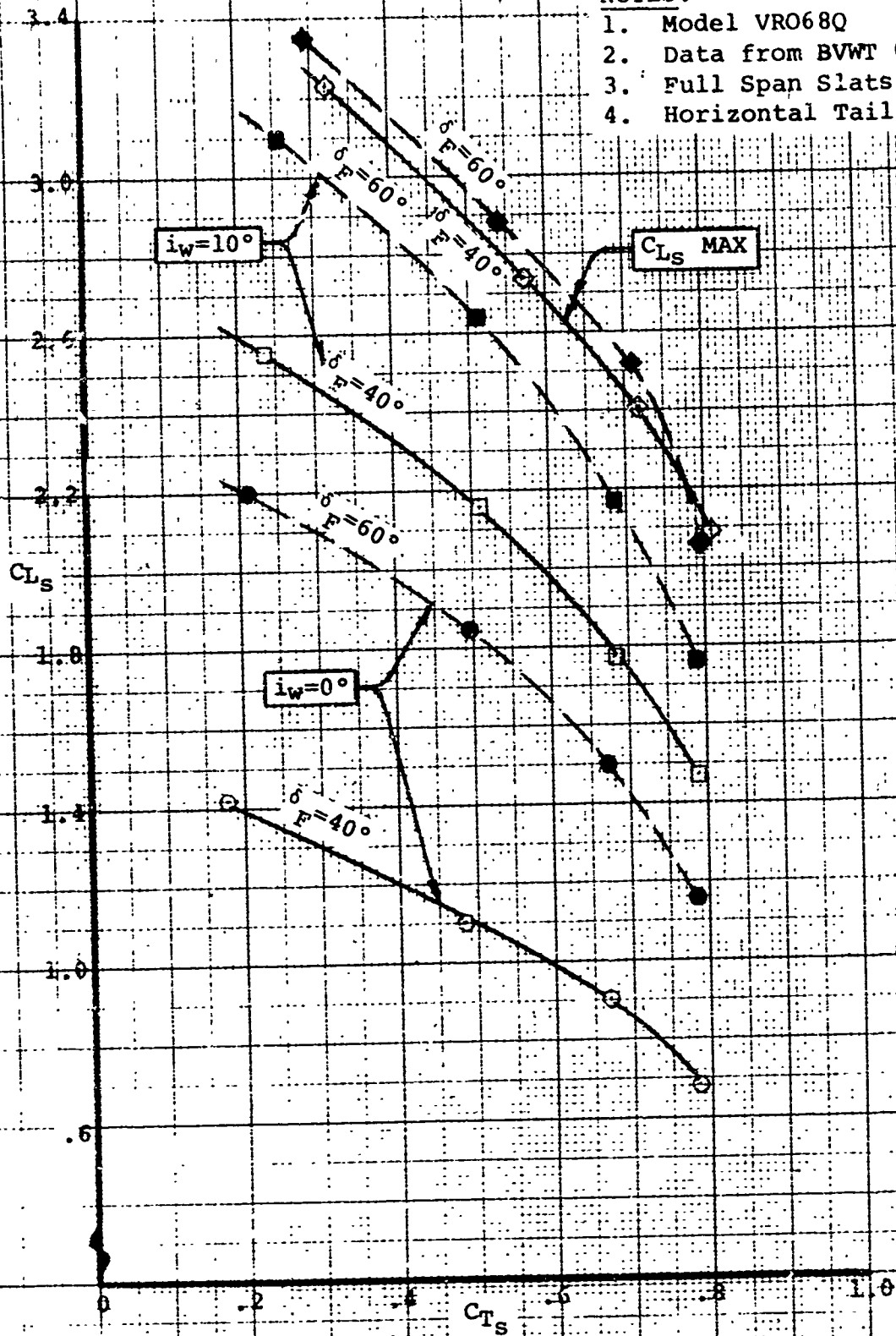
IN GROUND EFFECT LIFT COMPARISON
60° AND 40° FLAPS, h/D = 1.25

SHEET 222

EFFECT OF GROUND
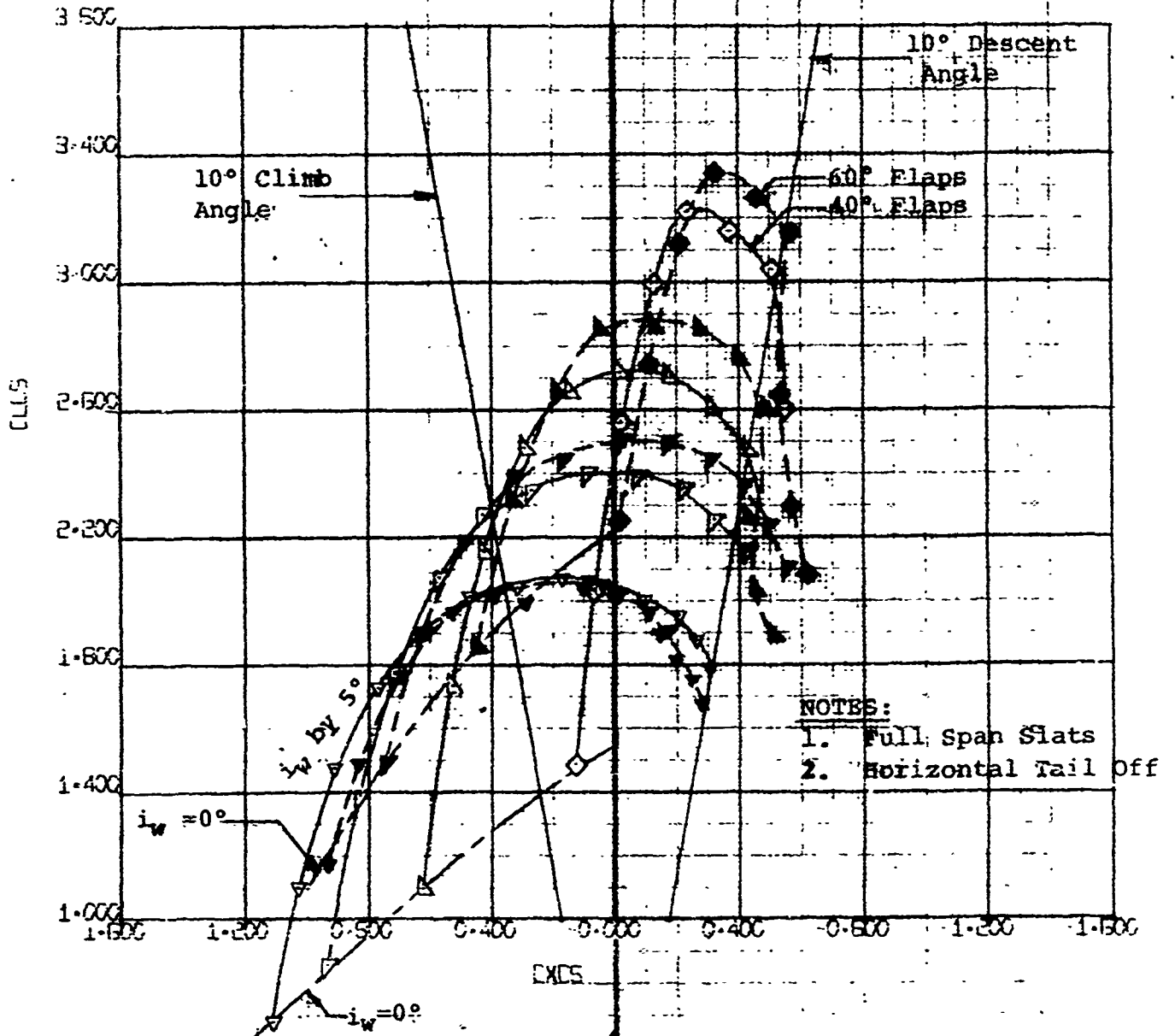ON LIFT CURVES
60° FLAPS; h/D=1.25

Open Symbols — Out-of-Ground Effect
Solid Symbols— In-Ground Effect
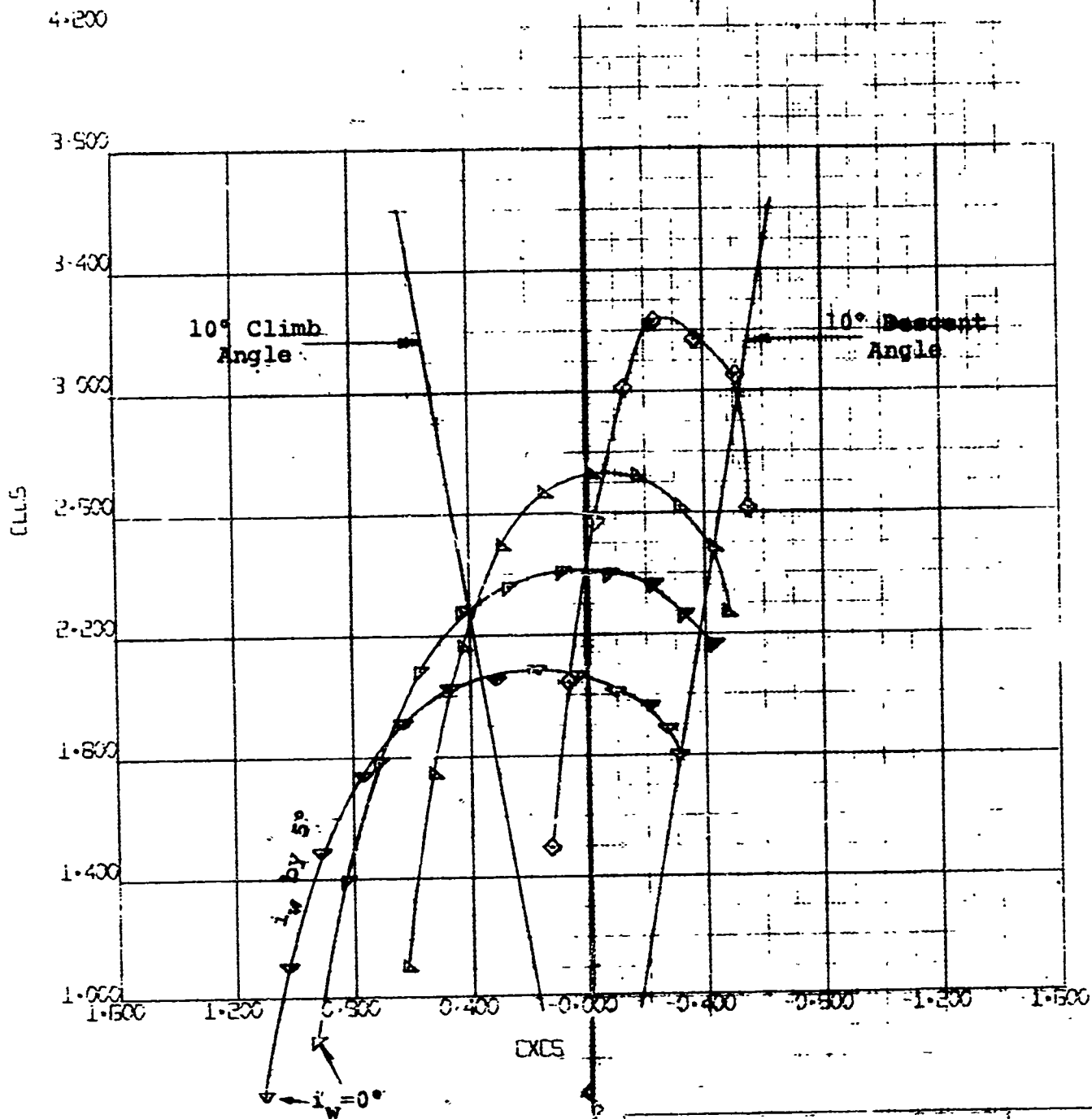
NOTES
1. Full Span Slats
2. Horizontal Tail Off

Sheet 223
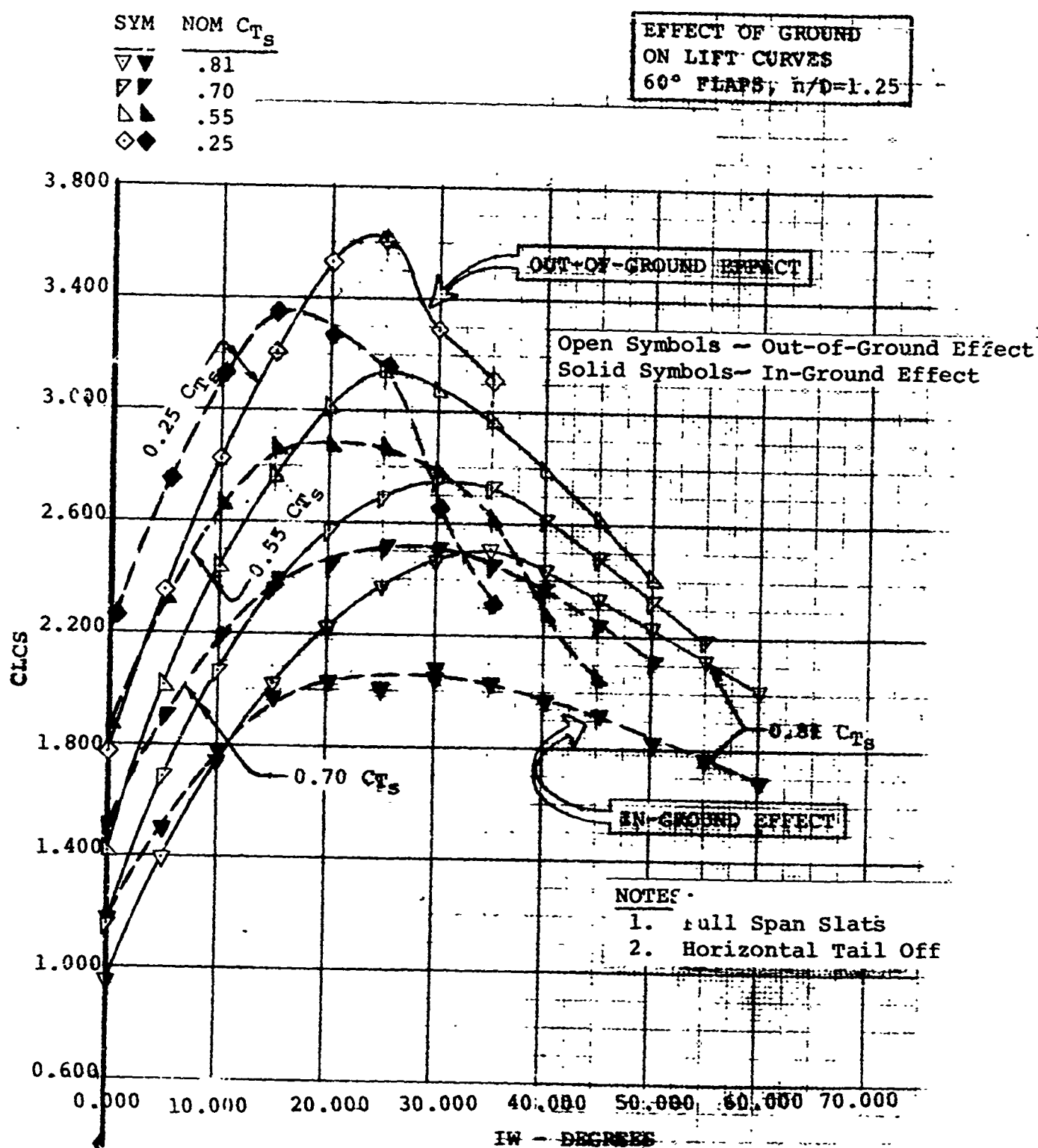
EFFECT OF GROUND ON FORCE POLARS 60° FLAPS, h/D=1.25

FOUR-PROP TILT WING MODEL VR06BC (FULL SPAN) CLCS VS· CXCS

NOTES:
1. Full Span Slats
2. Horizontal Tail Off

RUN	SYM	Δ		NOM C_{T_S}	Y
140	▽	---	2.7	.81	0°
141	▽	---	4.0	.70	
142	△	---	6.3	.55	
143	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0°$, $\delta_F = 40°$, h/D=1.25
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_S}	γ
140	▽	---	2.7	.81	0°
141	▽	---	4.0	.70	
142	△	---	6.3	.55	
143	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0°$, $\delta_F = 40°$, h/B=1.25
FULL SPAN SLATS

FOUR PROP TILT WING	BVWT
MODEL VRO68Q (FULL SPAN)	67
CLCS VS IW DEGREES	1/4/71

Sheet 226

RUN	SYM	$\triangle$	q	NOM C_{T_S}	γ
140	▽	---	2.7	.81	0°
141	▽	---	4.0	.70	
142	△	---	6.3	.55	
143	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0°$, $\delta_F = 40°$, h/D=1.25
FULL SPAN SLATS

FOUR PROP TILT WING   BVWT
MODEL VRO68O (FULL SPAN)   67
CMCS VS IW DEGREES   1/4/71

Sheet 227

Figure 155. Longitudinal Characteristics in Ground Effect, $\alpha_F = 0$, $\delta_F = 60°$, $h/D = 1.25$, Full Span Slats

RUN	SYM	$\triangle$	q	NOM C_{T_S}	Y
128	▽	---	2.7	.81	0°
129	▽	---	4.0	.70	
130	△	---	6.3	.55	
131	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0$, $\delta_F=60°$, $h/D=1.25$
FULL SPAN SLATS

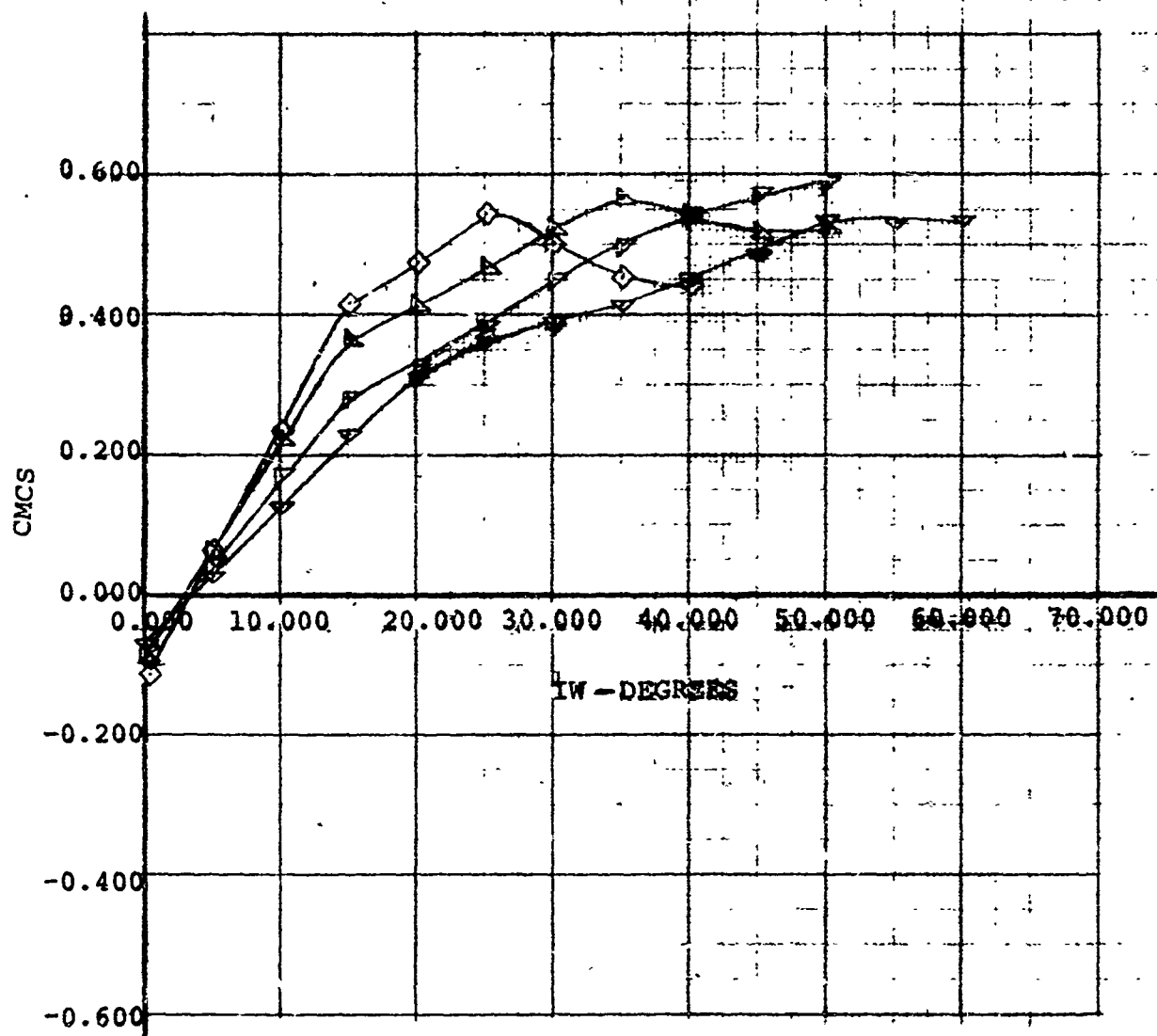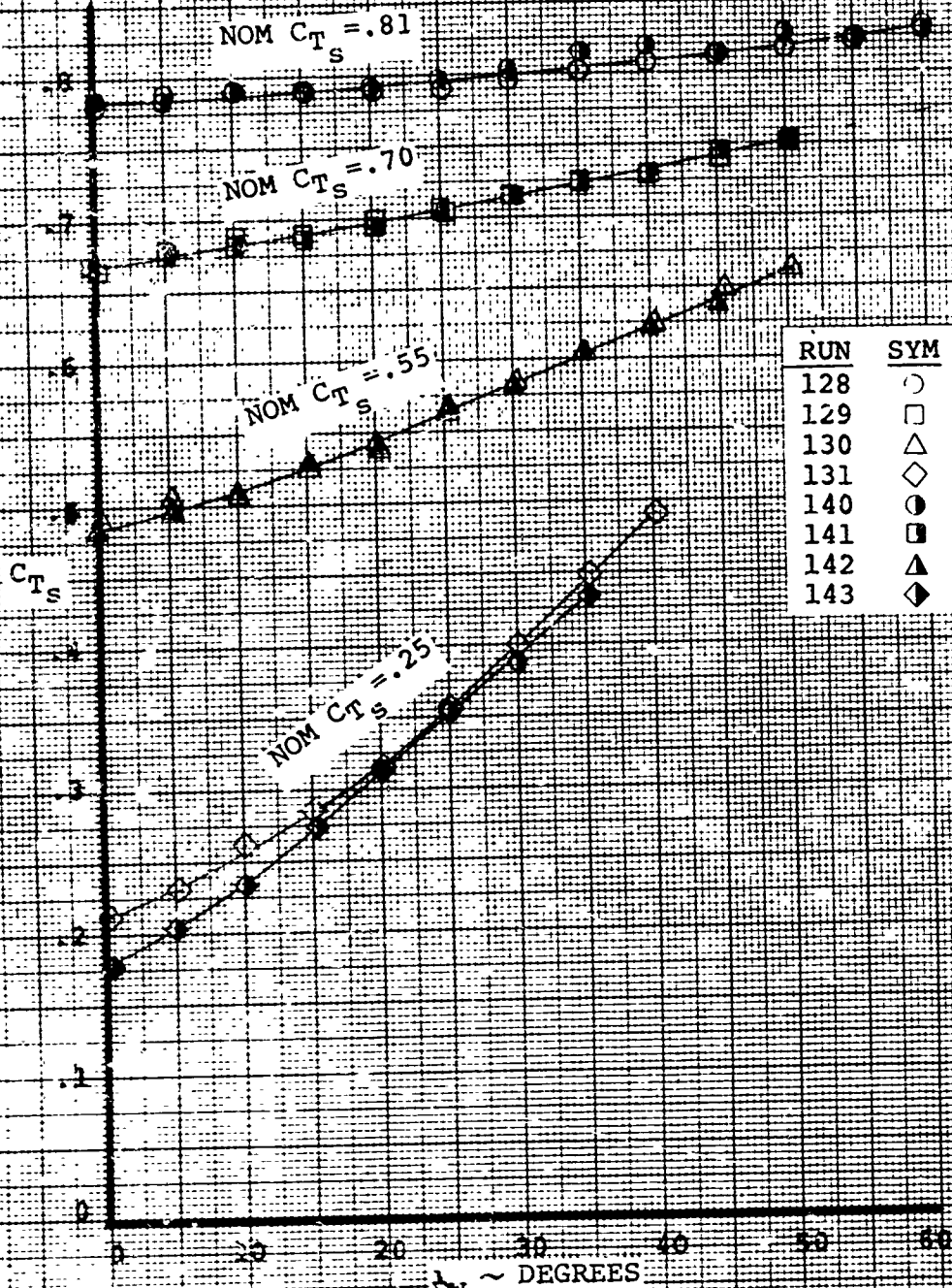| FOUR PROP TILT WING MODEL VR0680 (FULL SPAN) CLCS VS IW DEGREES | BVWT 67 12/17/70 |

Sheet 229

RUN	SYM	Δ	α	NOM C_{T_S}	γ
128	▽	---	2.7	.81	0°
129	▽	---	4.0	.70	
130	△	---	6.3	.55	
131	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0$, $\delta_F=60°$, h/D=1.25

FULL SPAN SLATS

FOUR PROP TILT WING	BVWT
MODEL VR068Q (FULL SPAN)	67
CMCS VS TW DEGREES	12/17/70

Sheet 230

NOTES:

1. Model VRO68.)
2. Data from BVWT 067
3. Full Span Slats
4. Horizontal Tail Off
5. Zero Fuselage Angle
6. Zero Cyclic Angle

NOM C_{T_s} = .81

NOM C_{T_s} = .70

NOM C_{T_s} = .55

C_{T_s}

NOM C_{T_s} = .25

RUN	SYM	δ_F	q
128	◌	60°	2.7
129	□		4.0
130	△		6.3
131	◇		10.6
140	◐	40°	2.7
141	◙		4.0
142	▲		6.3
143	◈		10.6

λ_w ~ DEGREES

VARIATION OF C_{T_s} WITH WING INCIDENCE
IN GROUND EFFECT (h/D=1.25)
4500 RPM

A check of the stabilizer effectiveness in the influence of the ground was desired as a part of the longitudinal investigation. This check was made at ' nominal slipstream thrust coefficient of 0.55, which corresponds to a full scale speed in the order of 60 knots. At this selected transition speed, the horizontal tail is still an important element of the low speed longitudinal control system.

Figures 159 through 161 present the three component results of the investigation. As in the previous I.G.E. longitudinal runs, the testing was performed via wing tilt sweeps with a level fuselage. The data, shown in slipstream coefficient form, was acquired with a 40° flap deflection and a cyclic hub operating speed of 4500 RPM.

A wing tilt angle of 11° was chosen for determining the horizontal tail pitching moment effectiveness from the results of the stabilizer angle runs presented in Figure 159. This tilt angle matches the condition selected for establishing the comparable O.G.E. horizontal tail effectiveness with 15° of wing tilt. The O.G.E. tail effectiveness data presented in Figure 71 of Reference 1 was obtained with 60° of flap and the collective set of hubs operating at 6800 RPM, settings different than those used during the in-ground effect testing. These differences, however, should have only a minor influence on the comparison.
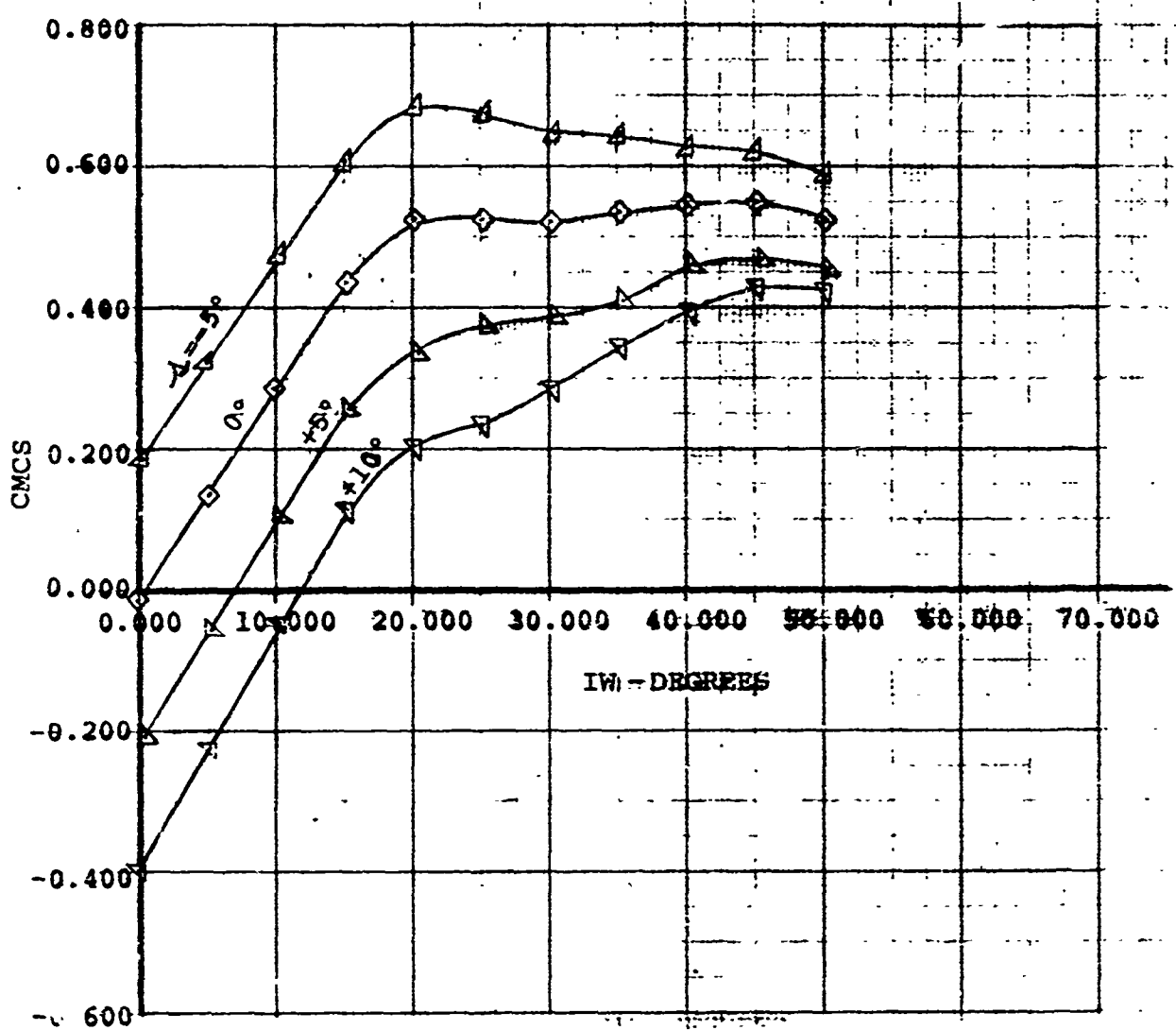
Figure 159 shows a $\Delta C_{M_S}/\Delta\Delta$ of .0345 for the in-ground effect condition, which compares to an out-of-ground effect value of .0338 at 0.52 C_{T_S} (Figure 71 of Reference 1). It would be expected that the horizontal tail effectiveness in-ground effect would increase to some extent by the combined influence of a larger q acting at the tail due to the ground plane suppressing the downward deflection of the wake, plus the increase in effective aspect ratio of the tail due to ground proximity.

RUN	SYM	Λ	q	NOM C_{T_S}	γ
144	△	+5°	6.3	.55	0°
146	◁	-5°			
147	▽	+10°			
148	◇	0°			

LONGITUDIAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_P=0$, $\delta_P=40°$, h/D=1.25
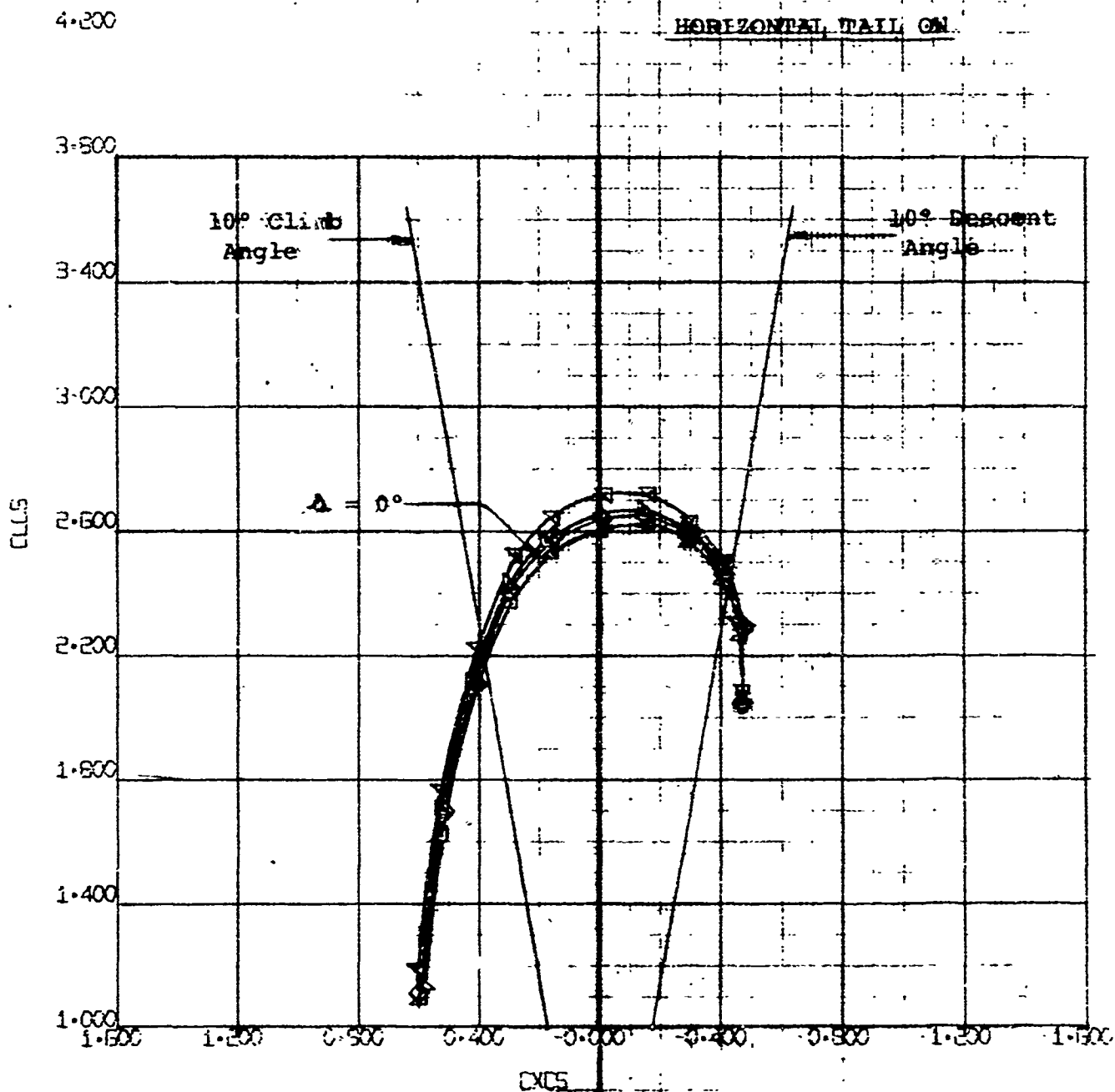FULL SPAN SLATS

HORIZONTAL TAIL ON
MOMENTS ABOUT WING PIVOT

FOUR PROP TILT WING	BVWT
MODEL VR068Q (FULL SPAN	67
CMCS VS IW DEGREES	1/4/71

RUN	SYM	△	α	NOM C_{T_s}	γ
144	▷	+5°	6.3	.55	0°
146	△	-5°			
147	◁	+10°			
148	◇	0°			

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\delta_F=0$, $\delta_F=40°$, $h/D=1.25$
FULL SPAN SLATS

HORIZONTAL TAIL ON

FOUR-PROP TILT WING	BVWT
MODEL VR06B0 (FULL SPAN)	67
CLLS VS. CXCS	1/4/71
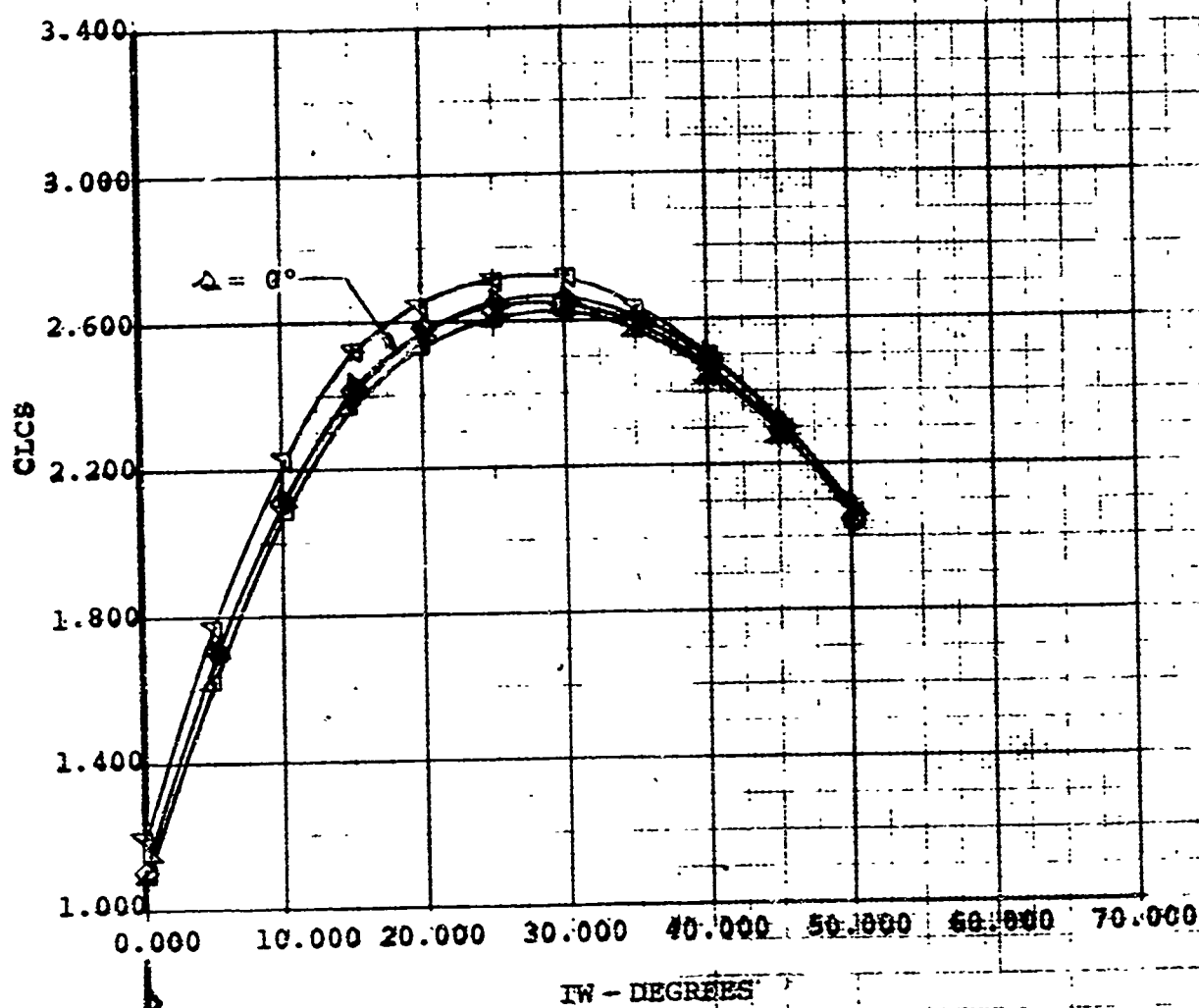
Figure 161. FOUR PROP TILT WING MODEL VRO68Q (FULL SPAN) CLCS VS IW DEGREES

6.7.2 Cyclic Pitch Control Effectiveness in Ground Effect

This section presents the 40° flap data acquired on cyclic
pitch control effectiveness in-ground effect during wing tilt
sweeps with the tail off. Cyclic pitch angles of +4° and -4°
were evaluated at the same slipstream thrust coefficients used
for the non-cyclic portion of the in-ground effect longitud-
inal investigation.

Figure 162 compares the cyclic control pitch capability in
terms of slipstream pitching moment per degree of cyclic, as
measured in-ground effect from the data included in this
report section, with that measured out-of-ground effect.
The O.G.E. data shown in Figure 162 was previously presented
in Figure 41. Even though the flap angles are different for
the two sets of data, 40° flaps for the I.G.E. data and 60°
flaps for the O.G.E. data, this configuration change should
not be of a magnitude that would materially alter the compari-
son. The variation of cyclic pitch effectiveness in-ground
effect, namely, the small increase with C_{T_s} at a constant
shaft angle (α_p) and the small decrease as the shaft angle is
increased, are the same trends exhibited by the 1/12th scale
isolated propeller data presented in Figure 41. Figure 162
shows that moving from an out-of-ground effect flight condi-
tion to a flight condition with the wheels 3 ft. off the
ground results in some reduction in the cyclic pitch effect-
iveness, but only by an average of 6%.

Three component data included in Figures 163 through 174 show
a direct comparison of the influence of positive and negative
cyclic on the pitching moment about the wing pivot, force
polars and lift curves. The slipstream format data is pres-
ented in sets of three plots in the order of increasing slip-
stream thrust coefficient. Data of the first set was
obtained at a nominal C_{T_s} of 0.25 and the last of the four
sets was obtained at a nominal C_{T_s} of 0.81.

Wing tilt sweeps with a level fuselage, that were performed in
lieu of the more conventional fuselage sweeps due to the low
ground height involved, do not allow a direct data comparison
to show the influence of ground effect on the aircraft
longitudinal stability. An examination of the in-ground
effect longitudinal pitching moment data does show, however,
that the effect of cyclic pitch inputs on the variation of
pitching moment with wing tilt angle to be small. A positive

236

cyclic pitch angle of +4° increases the tail-off pitching
moment slope by a small percentage (about 6%) and a negative
cyclic angle of -4° decreases it by a small percentage over
the C_{T_S} range evaluated with representative wing tilt angles.
This indication of the effect of cyclic pitch on longitudinal
stability is in the same direction with respect to angularity
of cyclic inputs and is of the same small magnitude as shown
in Figure 45 for the out-of-ground effect tail-off case.
Thus, it can be inferred that cyclic pitch control action
in-ground effect has the same influence on longitudinal
stability as recorded out-of-ground effect.

The in-ground effect lift curves interspersed in Figures 163
through 174 show the same trend observed out-of-ground effect,
in that positive cyclic increases the lift curve slope and
negative cyclic decreases it. Another item of interest is
the larger incremental loss in maximum lift with positive
cyclic at the lower thrust coefficients than was noted out-
of-ground effect.

Figures 175 through 180 include the same three component data
with cyclic pitch control applied, that was shown in Figures
163 through 174, but in a comparison of longitudinal character-
istics for a constant cyclic pitch angle. Figures 175 through
177 present the +4° cyclic data and Figures 178 through 180
present the -4° cyclic data.
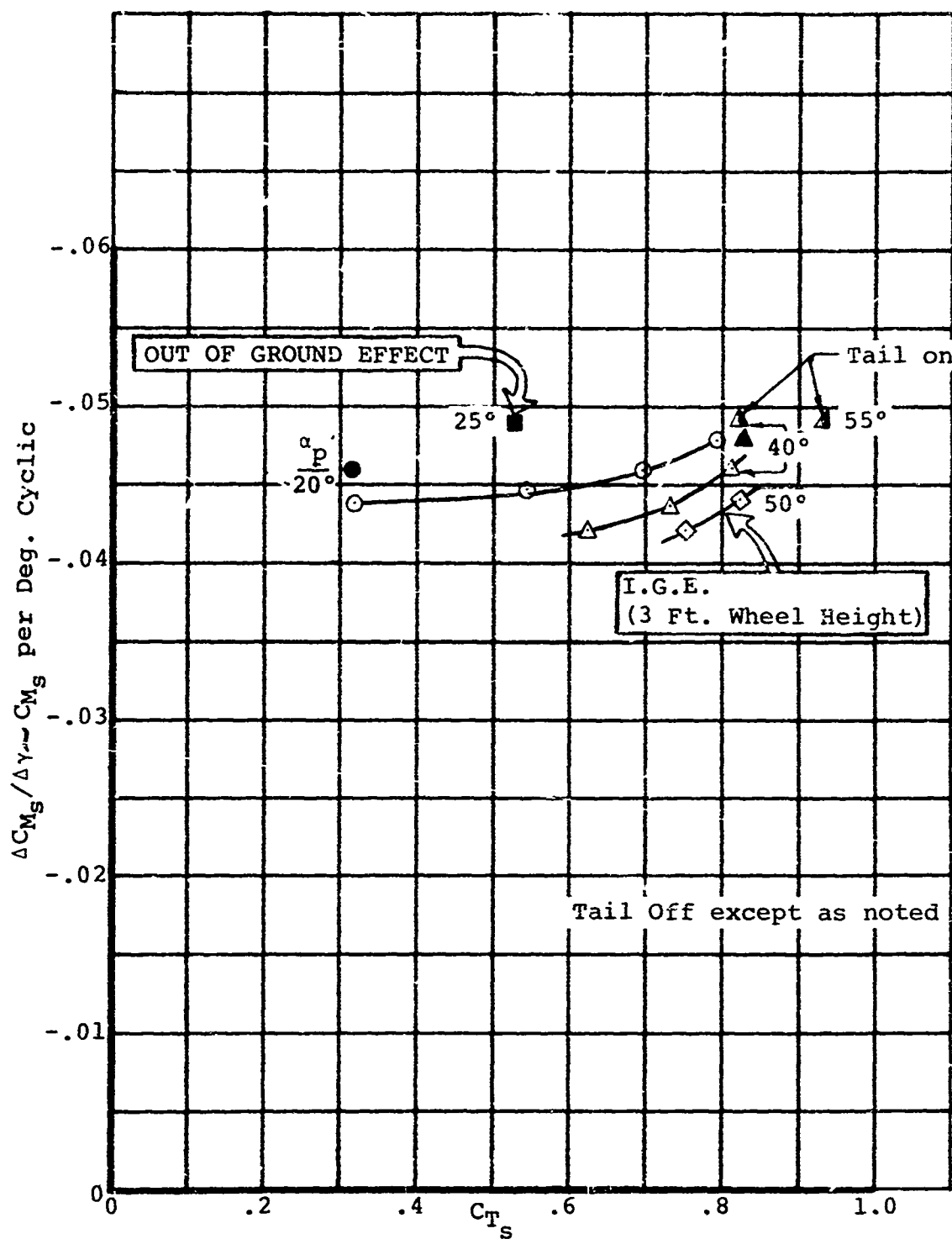
INFLUENCE OF GROUND EFFECT ON
CYCLIC PITCH EFFECTIVENESS

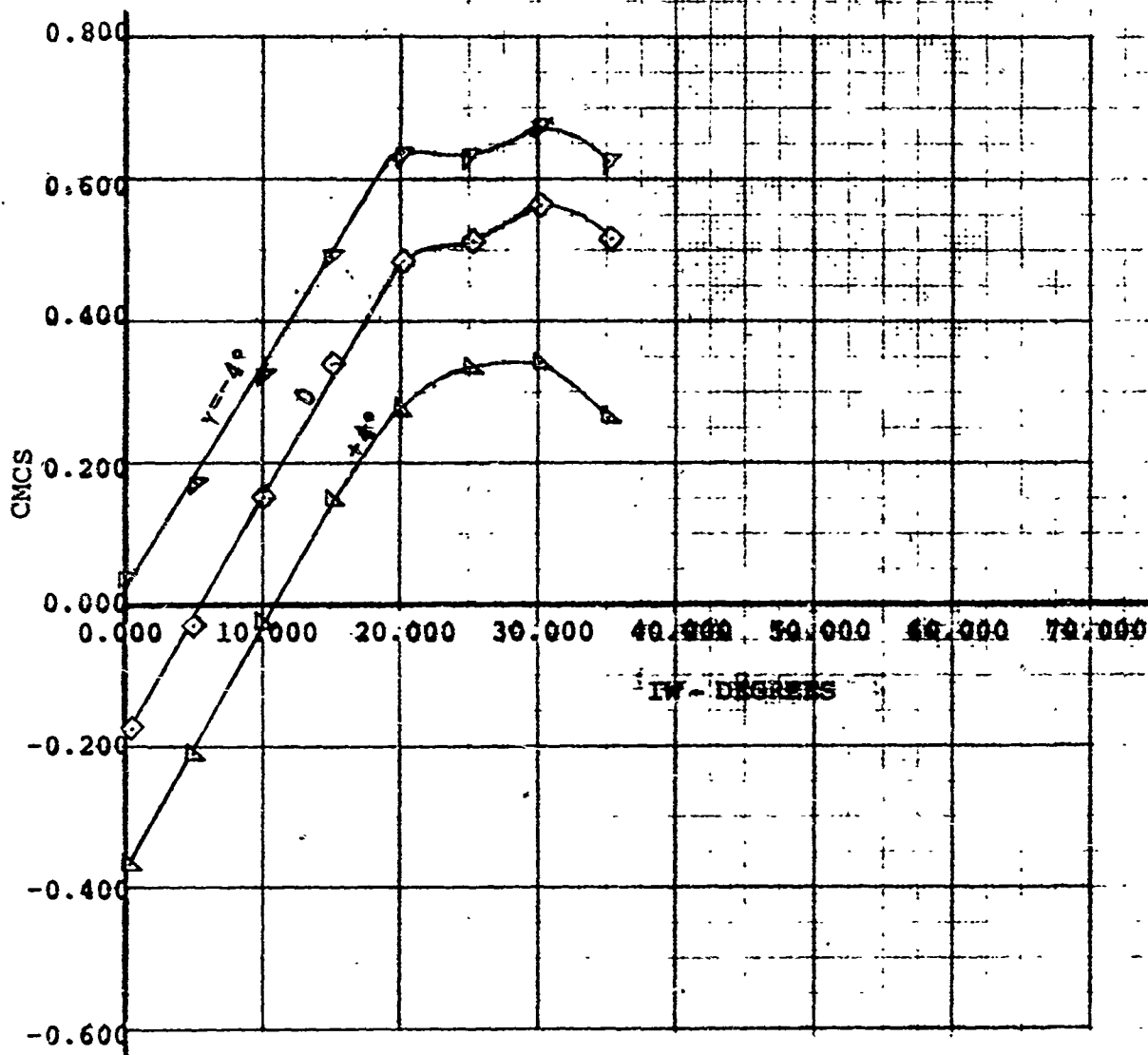LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$a_F = 0°$, $\delta_E = 40°$, h/D=1.25
FULL SPAN SLATS

With Cyclic Pitch Control

D170-10039-1
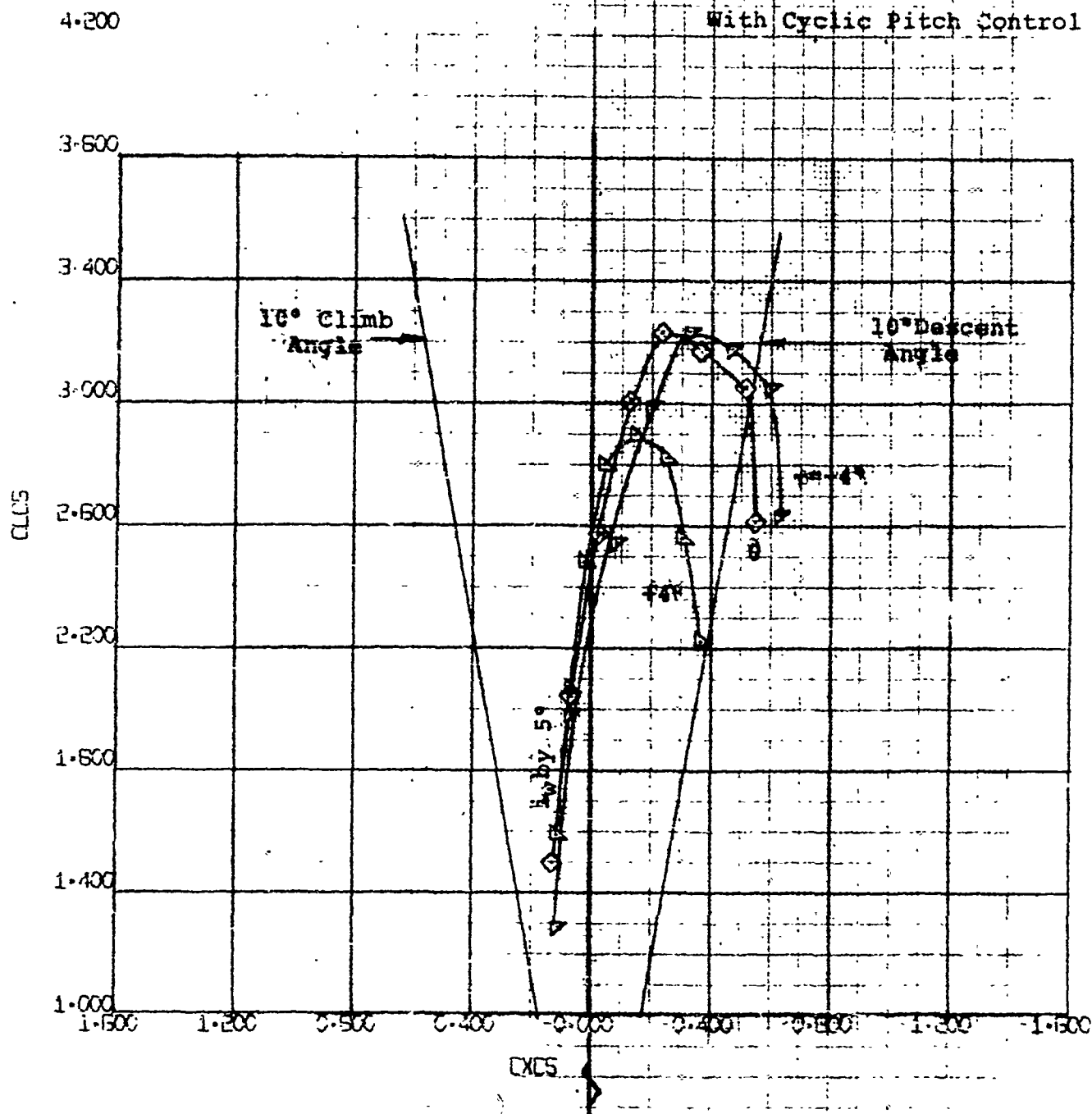Figure 163

RUN	SYM	A	q	NOM C_{T_S}	Y
143	◇	---	10.6	.25	0
152	△	---			+4°
156	▽	---			-4°

FOUR PROP TILT WING	BVWT
MODEL VRO58Q (FULL SPAN)	67
CMCS VS IW - DEGREES	1/4/71

239

RUN	SYM	Δ	q	NOM C_{T_s}	γ
143	◇	---	10.6	.25	0
152	△	---	↓	↓	+4°
156	▽	---	↓	↓	-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$a_F = 0°$, $\delta_F = 40°$, $h/D = 1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

FOUR PROP TILT WING   BVWT
MODEL VR068Q (FULL SPAN)   67
CLCS VS IW - DEGREES   1/4/71

241

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_T=0°$, $\delta_F=40°$, h/D=1.25
FULL SPAN SLATS

With Cyclic Pitch Control

D170-10039-1
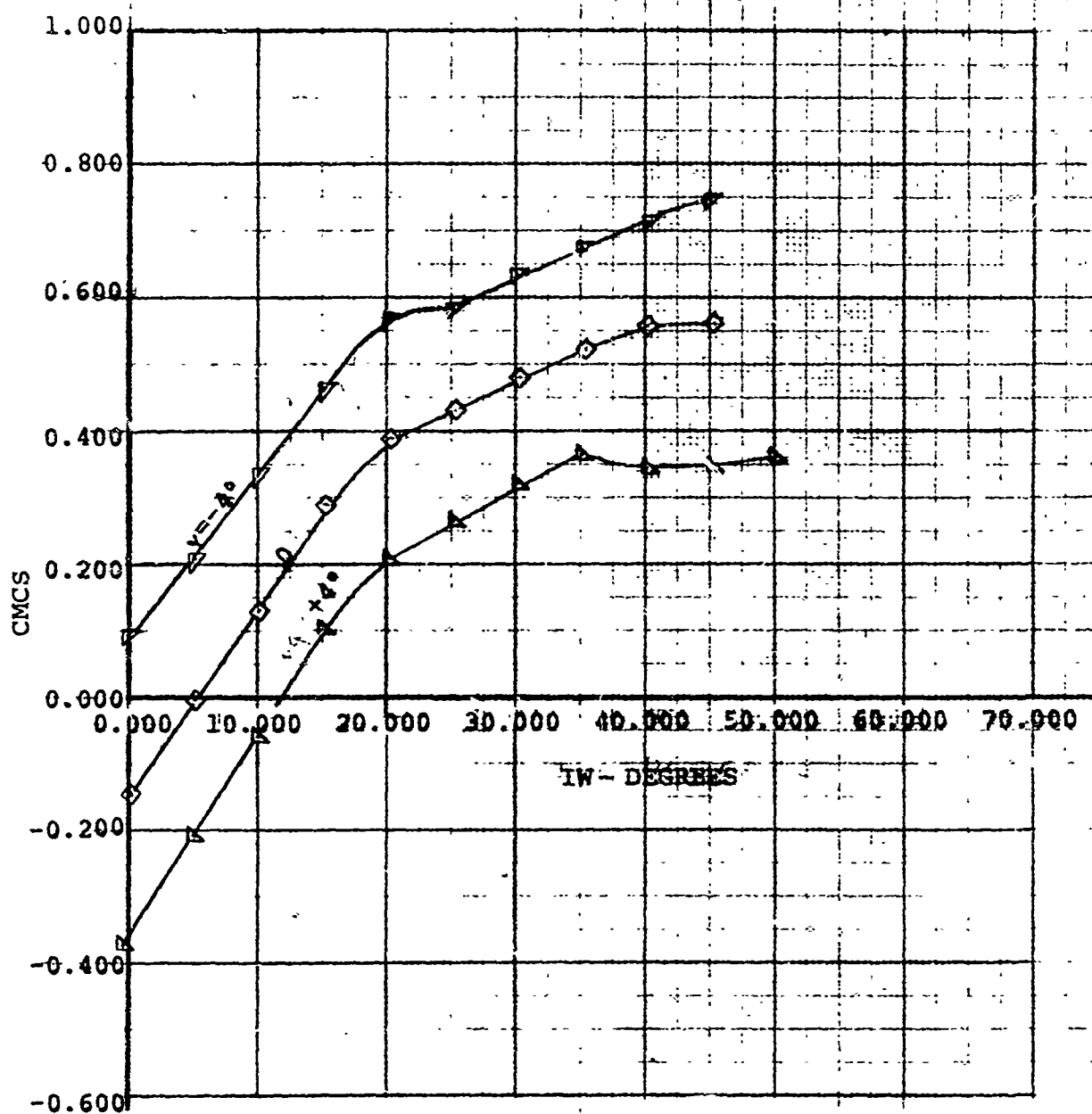Figure 166

RUN	SYM	δ	q	NOM C_{T_s}	Y
142	◇	---	6.3	.55	0°
151	△	---	↓	↓	+4°
155	▽	---			-4°

FOUR PROP TILT WING
MODEL VRO680 (FULL SPAN)
CMCS VS IW DEGREES

BVWT 67
1/4/71

242

RUN	SYM	△	q	NOM CT_s	Y
142	◇	---	6.3	.55	0°
151	⬠	---	↓	↓	+4°
155	▽	---			-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$a_F = 0°, 6_F = 40°, h/D = 1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

4·200

3·800

3·400

3·000

10°Climb
Angle

10°Descent
Angle

2·600

CLLS

2·200

1·800

i_w by 5°

1·400

1·000

-1·600 -1·200 -0·800 -0·400 0·000 -0·400 -0·800 -1·200 -1·600

CXCS

FOUR-PROP TILT WING	BVWT
MODEL VR-55B (FULL SPAN)	67
CLCS VS· CXCS	1/ 4/71

SHEET 243

FOUR PROP TILT WING MODEL VRO68Q (FULL SPAN)

CLCS VS TW - DEGREES

RUN	SYM	Δ	q	NOM C_{T_B}	γ
141	◇	---	4.0	.70	0°
150	△	---	↓	↓	+4°
154	▽	---			-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$i_F=0°$, $\delta_F=60°$, $h/D=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

CMCS

IW- DEGREES

FOUR PROP TILT WING	BVWT
MODEL VRO68Q (FULL SPAN)	67
CMCS VS IW DEGREES	1/4/71

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\delta_F=0°$, $\delta_F=40°$, $h/D=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

RUN	SYM	h	α	NOM C_{T_S}	γ
141	◇	---	4.0	.70	0°
150	△	---			±4°
154	▽	---			−4°

4·200

3·800

3·400

3·000

10° Climb Angle

10° Descent Angle

2·600

CLCS

2·200

1·800

1·400

1·000

1·600   1·200   0·800   0·400   0·000   −0·400   −0·800   −1·200   −1·600

CXCS

FOUR-PROP TILT WING	BVWT
MODEL VRC680 (FULL SPAN)	67
CLCS VS. CXCS	1/4 71

RUN	SYM	Δ	q	NOM-C_{T_S}	γ
141	◇	---	4.0	.70	0°
150	△	---	↓	↓	+4°
154	▽	---	↓	↓	-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$n_F=0°$, $\delta_m=40°$, h/D=1.25
FULL SPAN SLATS

With Cyclic Pitch Control

FOUR PROP TILT WING
MODEL VR0680 (FULL SPAN)

CLCS VS IW DEGREES

BVWT 67

1/4/71

247

D170-10039-1
Figure 172

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0°$, $\delta_F=40°$, $h/D=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

RUN	SYM	α	q	NOM C_{T_S}	Y
140	◇	---	2.7	.81	0
149	◁	---			+4°
153	▽	---			-4°

FOUR PROP TILT WING
MODEL VR068Q (FULL SPAN)
CMCS VS IW—DEGREES
BVWT 67
1/4/71

248

RUN	SYM	$\triangle$	q	NOM C_{T_S}	Y
140	◇	---	2.7	.81	0
149	△	---			+4°
153	▽	----			-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0°, \delta_F=40°, h/D=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

CLCS

4·200

3·800

3·400

3·000

10°Climb
Angle

10°Descent
Angle

2·600

2·200

1·800

1·400

1·000

BY 5°

$\gamma=0$

1·800 1·200 0·800 0·400 0·200 0·400 0·800 1·200 1·500

CXCS

FOUR-PROP TILT WING	BVWT
MODEL VROSSO (FULL SPAN)	67
CLCS VS· CXCS	1/ 4/71

SHEET 249

RUN	SYM	Δ	q	NOM CT_S	γ
140	◇	---	2.7	.81	0
149	△	---	↓	↓	+4°
153	▽	---	↓	↓	-4°

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_p=0°$, $\delta_F=40°$, $h/D=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

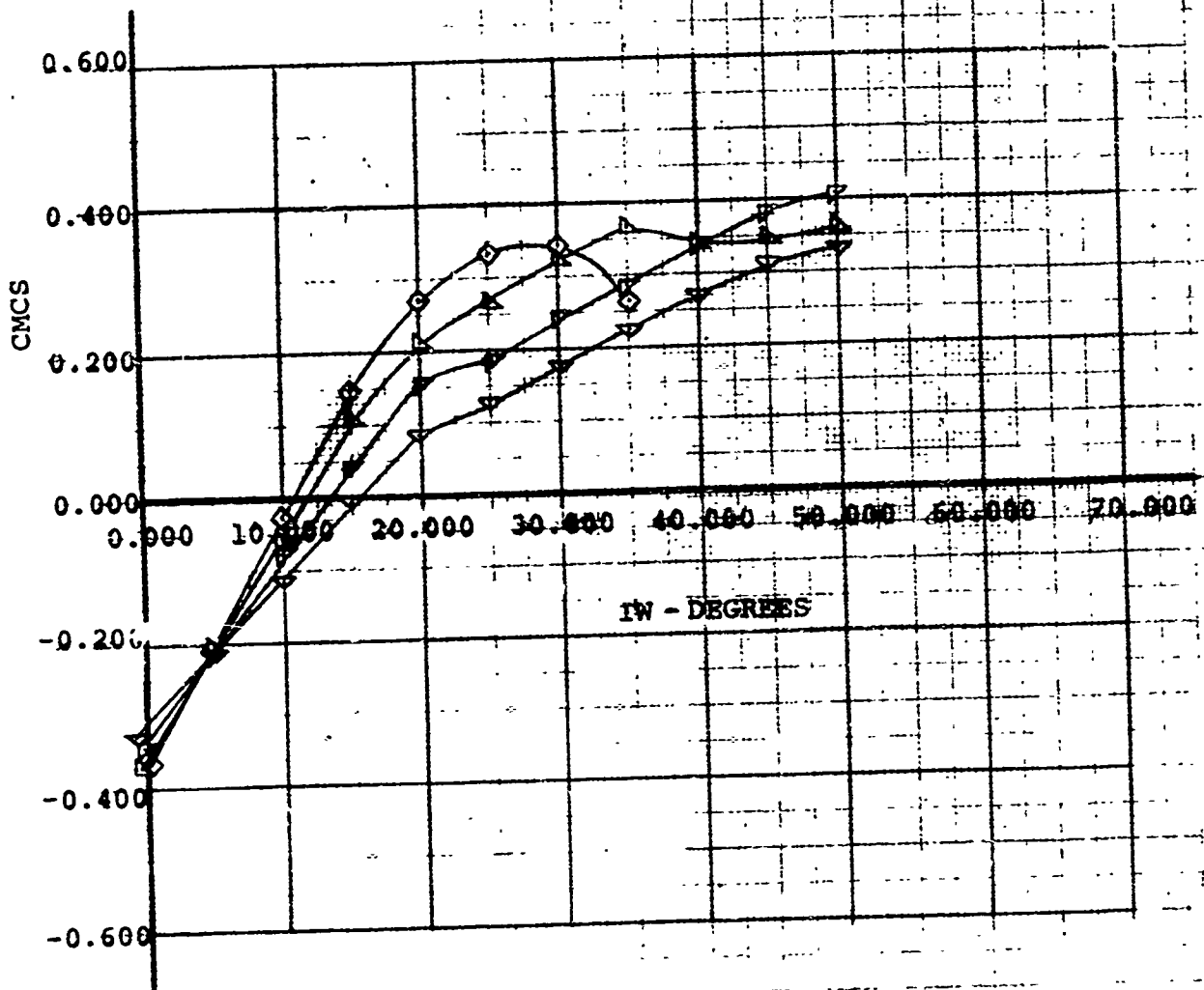|	FOUR PROP TILT WING	BVWT
	MODEL VRO68Q (FULL SPAN)	67
	CLCS VS IW DEGREES	1/4/7

250

RUN	SYM	△	σ	NOM C_T	Y
149	▽	---	2.7	.81	+4°
150	▼	---	4.0	.70	↓
151	▲	---	6.3	.55	
152	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0°$ $\delta_F = 40°$, h/D=1.25
FULL SPAN SLATS

CMCS

IW - DEGREES

FOUR PROP TILT WING	BVWT
MODEL VROGWO (FULL SPAN)	67
CMCS VS IW DEGREES	1/4/71

251

RUN	SYM	$\triangle$	q	NOM C_{T_s}	γ
149	▽	---	2.7	.81	+4°
150	◁	---	4.0	.70	
151	△	---	6.3	.58	
152	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0°$, $\delta_R = 40°$, $h/D = 1.25$
FULL SPAN SLATS

FOUR-PROP TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CLCS VS. CXCS	1/4/7.

RUN	SYM	A	σ	NOM C_{T_S}	γ
149	▽	---	2.7	.81	+4°
150	▷	---	4.0	.70	
151	△	---	6.3	.56	
152	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0°$, $\theta_F=40°$, h/D=1.25
FULL SPAN SLATS

γ=+4°

CLCS

IW — DEGREES

FOUR PROP TILT WING	BVWT
MODEL VR068Q (FULL SPAN)	67
CLCS vs IW DEGREES	1/4/71

RUN	SYM	$\perp$	q	NOM C_{T_E}	Y
153	▽	---	2.7	.81	-4
154	▽	---	4.0	.70	
155	△	---	6.3	.55	
156	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0$, $\delta_F = 40°$, h/D=1.25
FULL SPAN SLATS

CMCS

IW - DEGREES

FOUR PROP TILT WING	BVWT
MODEL VRO68Q (FULL SPAN)	67
CMCS VS IW - DEGREES	1/4/71

254

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F=0$, $\delta_F=40°$, $h/D=1.25$
FULL SPAN SLATS

RUN	SYM	Δ	q	NOM C_{T_s}
153	▽	---	2.7	.81
154	▽	---	4.0	.70
155	△	---	6.3	.56
156	◇	---	10.6	.25

10° Climb Angle

10° Descent Angle

i_W by 5°

$i_W=0°$

CLCS

CXCS

FOUR-PROP TILT WING.
MODEL VR0680 (FULL SPAN)
CLCS VS. CXCS

BVWT
67

1/ 4/71

RUN	SYM	Δ	q	NOM C_{T_S}	γ
153	▽	---	2.7	.81	-4
154	▽	---	4.0	.70	
155	△	---	6.3	.55	
156	◇	---	10.6	.25	

LONGITUDINAL CHARACTERISTICS
IN GROUND EFFECT
$\alpha_F = 0$, $\delta_F = 40°$, h/D=1.25
FULL SPAN SLATS

$\gamma = -4$

CLCS vs IW — DEGREES

CLCS (vertical axis): 3.800, 3.400, 3.000, 2.600, 2.200, 1.800, 1.400, 1.000, 600

IW — DEGREES (horizontal axis): 0.000, 10.000, 20.000, 30.000, 40.000, 50.000, 60.000, 70.000

FOUR PROP TILT WING	BVWT
MODEL VRO68C (FULL SPAN)	67
CLCS VS IW – DEGREES	1/4/71

6.8 LATERAL/DIRECTIONAL STABILITY IN GROUND EFFECT INCLUDING CYCLIC PITCH EFFECTS

The same test configuration utilized for longitudinal in-ground effect testing was also used for the in-ground effect lateral/ directional testing, namely, the Boeing-Vertol moving ground plane and a level fuselage with a scaled ground height of 1.25 h/D representing the aircraft with the wheels 3 ft. from the ground. Slats were extended and the flaps were deflected to the 60° setting.

Empennage-on yaw sweeps were performed with and without cyclic pitch inputs at typical combinations of wing tilt angle and slipstream thrust coefficient, through transition, that were previously evaluated during the out-of-ground effect phase of the lateral/directional testing. All of these yaw runs were conducted with the horizontal tail mounted high on the fin per Figure 15 and with a constant propeller speed of 4500 RPM. As in the out-of-ground effect testing, the model was off-set from the vertical centerline of the test section so that a yaw angular range of 8° (nose right) and -24° (nose left) could be achieved.

All moments obtained during the in-ground effect lateral/ directional runs were transferred to the same wing down mid center of gravity location and movement with wing tilt angle as used for the O.G.E. data analysis. Again, the body axis system was used for the data presentations.

The results of the in-ground effect lateral/directional stability investigation with empennage on are included in Figure 181 as slipstream stability derivatives. This plot presents the slipstream yawing moment, rolling moment, and side force derivatives (with respect to the sideslip angle, β) as a function of the slipstream thrust coefficient. As in the analysis of the lateral/directional data obtained out-of-ground effect, all slopes plotted in Figure 181 were read over a yaw angle range that did not exceed $\pm 8°$ and the C_{T_S} values were those prevailing at zero yaw angle. Also shown in this figure is the comparable out-of-ground data (dashed lines) obtained with zero cyclic angles. This empennage on O.G.E. data was originally presented in Figure 96.

With 15° of wing tilt and zero cyclic the primary influence of ground effect was to incrementally reduce the directional stability over the .29 to .69 C_{T_S} range evaluated. It is not discernible whether the decrease in directional stability is primarily caused by an increase in the basic tail-off instability or a loss in vertical tail effectiveness since only tail-on data was acquired. Increasing the wing tilt angle to 30° exhibited the same trend up to a C_{T_S} of 0.76, which corresponds to a full-scale speed of approximately 45 knots.

A possible source of a portion of the reduction in I.G.E. directional stability with 15° of wing tilt is the associated increase in slipstream lift coefficient over that developed O.G.E with the same empennage-on model configuration. This is shown in Figure 182 which compares the slipstream lift coefficients produced at zero yaw angle during the I.G.E. lateral/ directional testing, with the lift coefficients produced during the comparable O.G.E lateral/directional testing.

It can be noted in Figure 181 that in the C_{T_S} range up to 0.76 the effect of a positive cyclic input is similar to the result recorded out-of-ground effect in that the chang s in lateral/directional stability are small. Negative cyclic has a more pronounced influence in-ground effect, increasing both the directional stability and dihedral effect by larger increments than for the out-of-ground effect case.
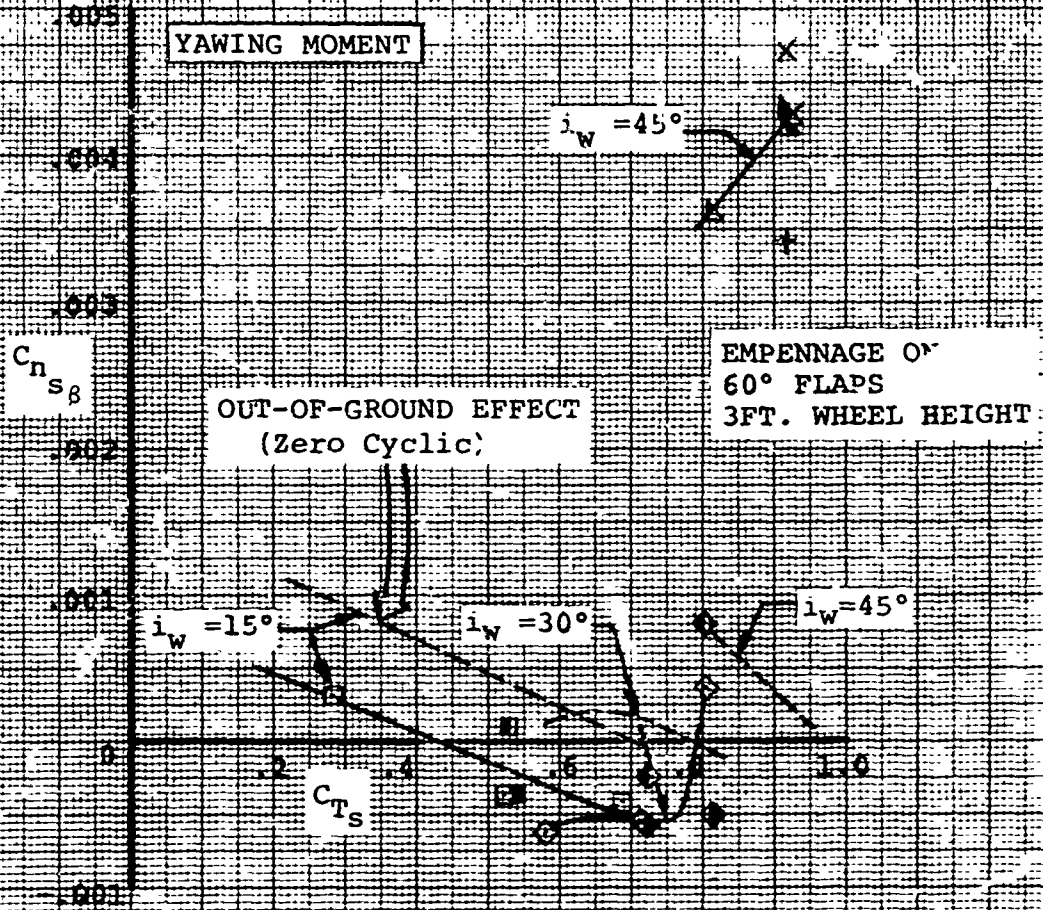
A condition was encountered at 0.8 C_{T_S} that is identified as ground recirculation. At this test point an increase in directional stability, dihedral effect, and side force were measured with zero cyclic pitch. The change to these derivatives that occurred with cyclic pitch inputs, a negative angle further increasing the derivatives and a positive angle reducing the derivatives to a level consistent with the lower C_{T_S} data, leads to the conclusion that at this condition ground recirculation was in the initial stages of formation. The application of negative cyclic, which effectively increases the propeller shaft angle and thus the angle of attack of the total lifting system, induced a stronger case of ground recirculation. Positive cyclic lowered the effective angle of attack sufficiently to completely relieve the model from ground recirculation. A lower flap angle would produce the same result. A question exists as to whether the low tunnel q prevailing at this test point (2.7 lb/ft^2) was a factor in precipitating ground recirculation in this particular condition.

258

At the high C_{T_S} values of .81 to .92 investigated with 45°
of wing tilt, ground recirculation was well developed as
evidenced by a large increase in directional stability and
dihedral effect over that recorded out-of-ground effect. The
magnitude of these stability changes was probably influenced
by the wing configuration; a tapered wing with straight lead-
ing edge and swept forward flap hingeline. It would appear
from the direction of these forces and moments, nose-right
yawing moment, right wing-up rolling moment and left side
force that the up-wind wing and fuselage side tended to "funnel"
the air into the wing/body junctio.. and the air on the down-
wind wing side was partially diverted spanwise and around the
wing tip.

Figures 183 through 191 include the basic three component yaw
data with zero cyclic pitch that was acquired during the in-
ground effect testing. The data, presented in sets of yawing
moment, rolling moment and side force plots, is arranged in
order of increasing wing tilt angle from 15° to 30° to 45°.
Each set of plots depicts the data for the series of slip-
stream thrust coefficients examined at a particular wing tilt
angle.

The three component data obtained during the I G.E. lateral/
directional testing with cyclic pitch inputs is presented in
Figures 192 through 203. This data is arranged in a manner
similar to that used in presenting the zero cyclic data, with
each set of three plots depicting the data obtained from the
various cyclic pitch runs performed with a particular wing
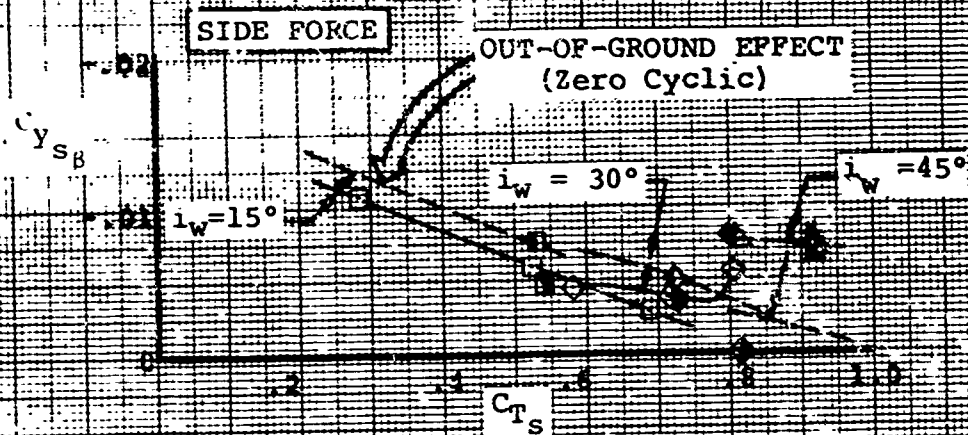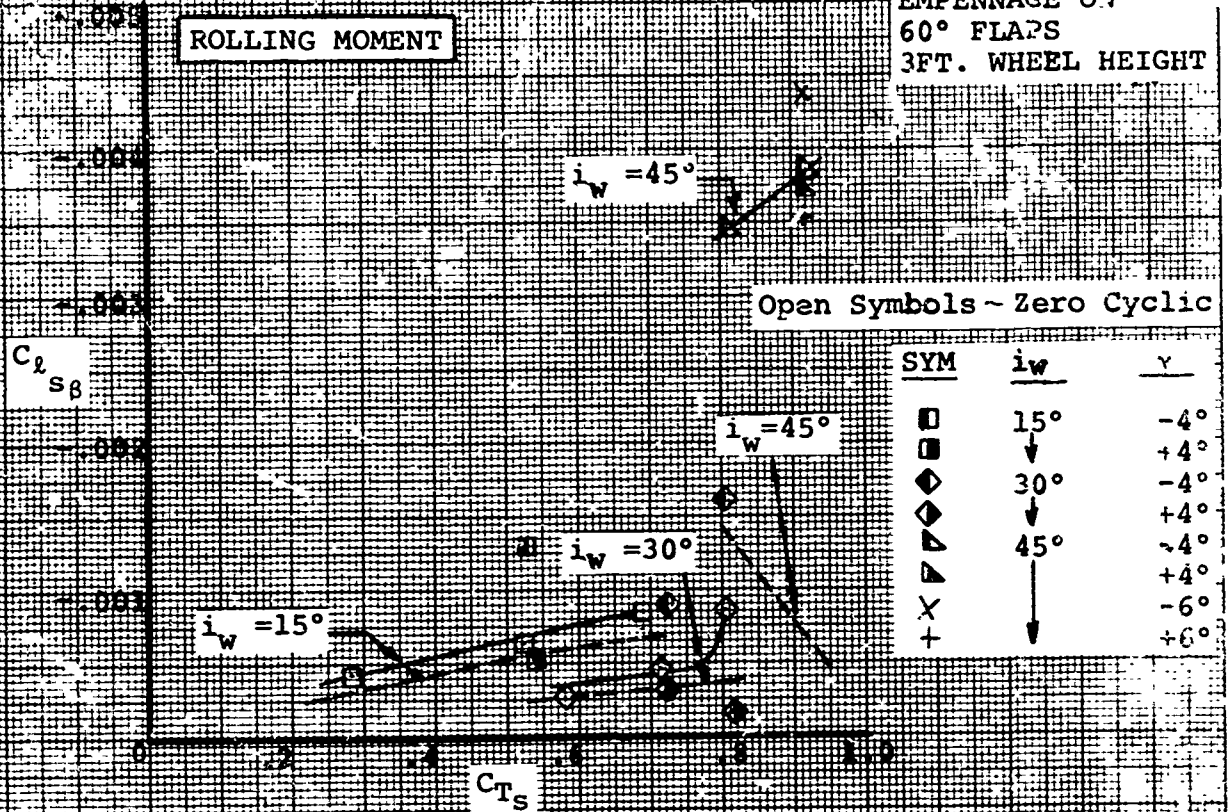tilt angle.

259

EFFECT OF CYCLIC PITCH ON
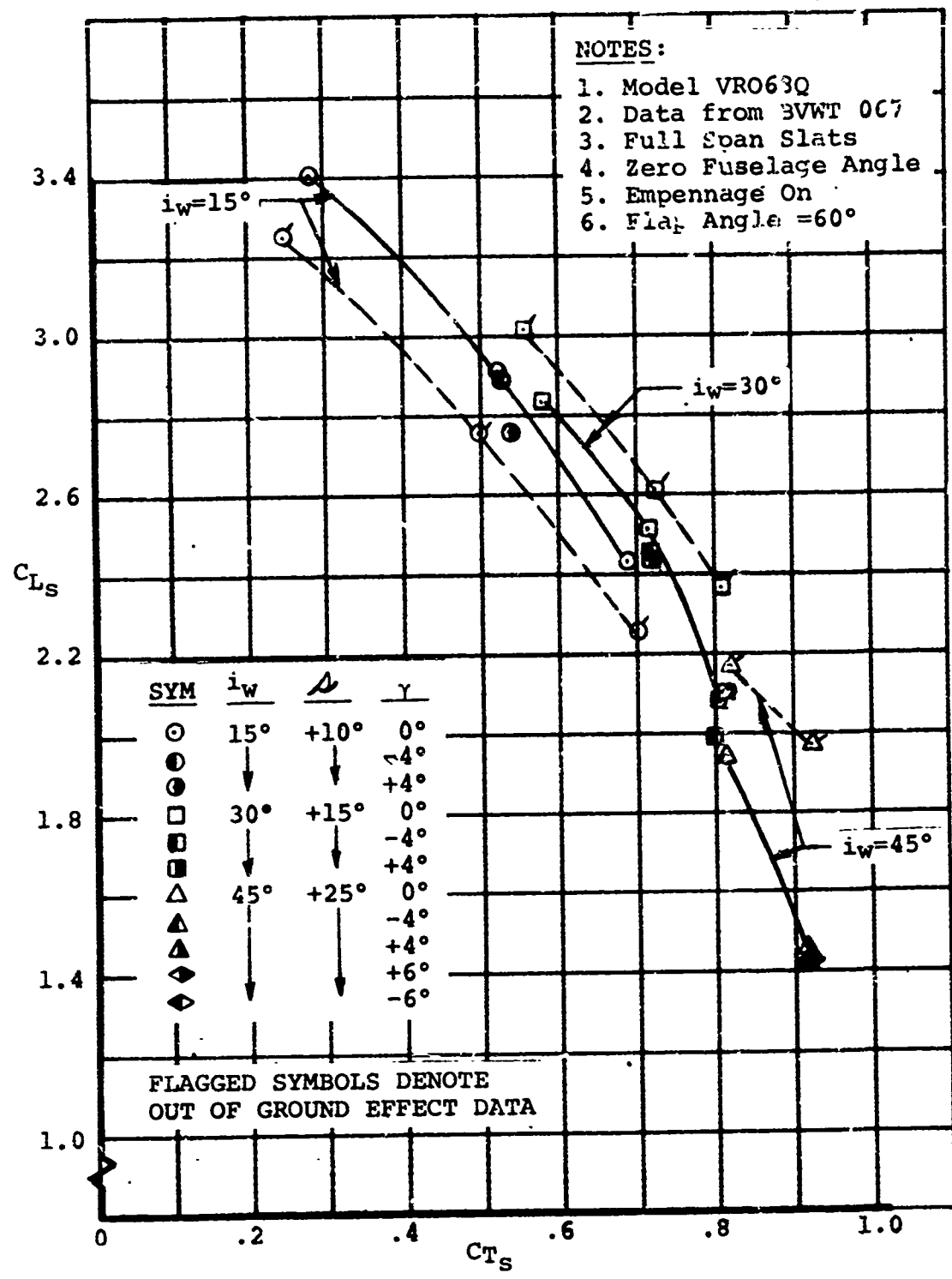LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT

YAWING MOMENT

$i_w = 45°$

$C_{n_{s_\beta}}$

EMPENNAGE ON
60° FLAPS
3FT. WHEEL HEIGHT

OUT-OF-GROUND EFFECT
(Zero Cyclic)

$i_w = 15°$ $i_w = 30°$ $i_w = 45°$

C_{T_s}

Open Symbols ~ Zero Cyclic

SYM	i_w	γ
▯	15°	-4°
▮	↓	+4°
◈	30°	-4°
◆	↓	+.°
◣	45°	-4°
◮	↓	+4°
X	↓	-6°
+	↓	+6°

EFFECT OF CYCLIC PITCH ON
LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT

ROLLING MOMENT

EMPENNAGE ON
60° FLAPS
3FT. WHEEL HEIGHT

Open Symbols ~ Zero Cyclic

$C_{\ell_{s_\beta}}$

$i_w = 45°$

$i_w = 45°$

$i_w = 30°$

$i_w = 15°$

SYM	i_w	Y
◻	15°	-4°
◼	↓	+4°
◆	30°	-4°
◈	↓	+4°
◮	45°	-4°
◮		+4°
X		-6°
+	↓	+6°

C_{T_s}

SIDE FORCE

OUT-OF-GROUND EFFECT
(Zero Cyclic)

$C_{y_{s_\beta}}$

$i_w = 30°$

$i_w = 45°$

$i_w = 15°$

C_{T_s}

SLIPSTREAM C_L VARIATION DURING YAW TESTS
IN GROUND EFFECT, h/D =1.25

RUN	SYM	δ	q	NOM C_{T_s}	γ
132	▽	+10°	4.0	.70	0°
133	△	↓	6.3	.50	↓
134	▶	↓	10.6	.25	↓

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=15°, δ_F=60°, h/D=1.25
FULL SPAN SLATS

BETA - ANGLE OF SIDESLIP

CYMCS
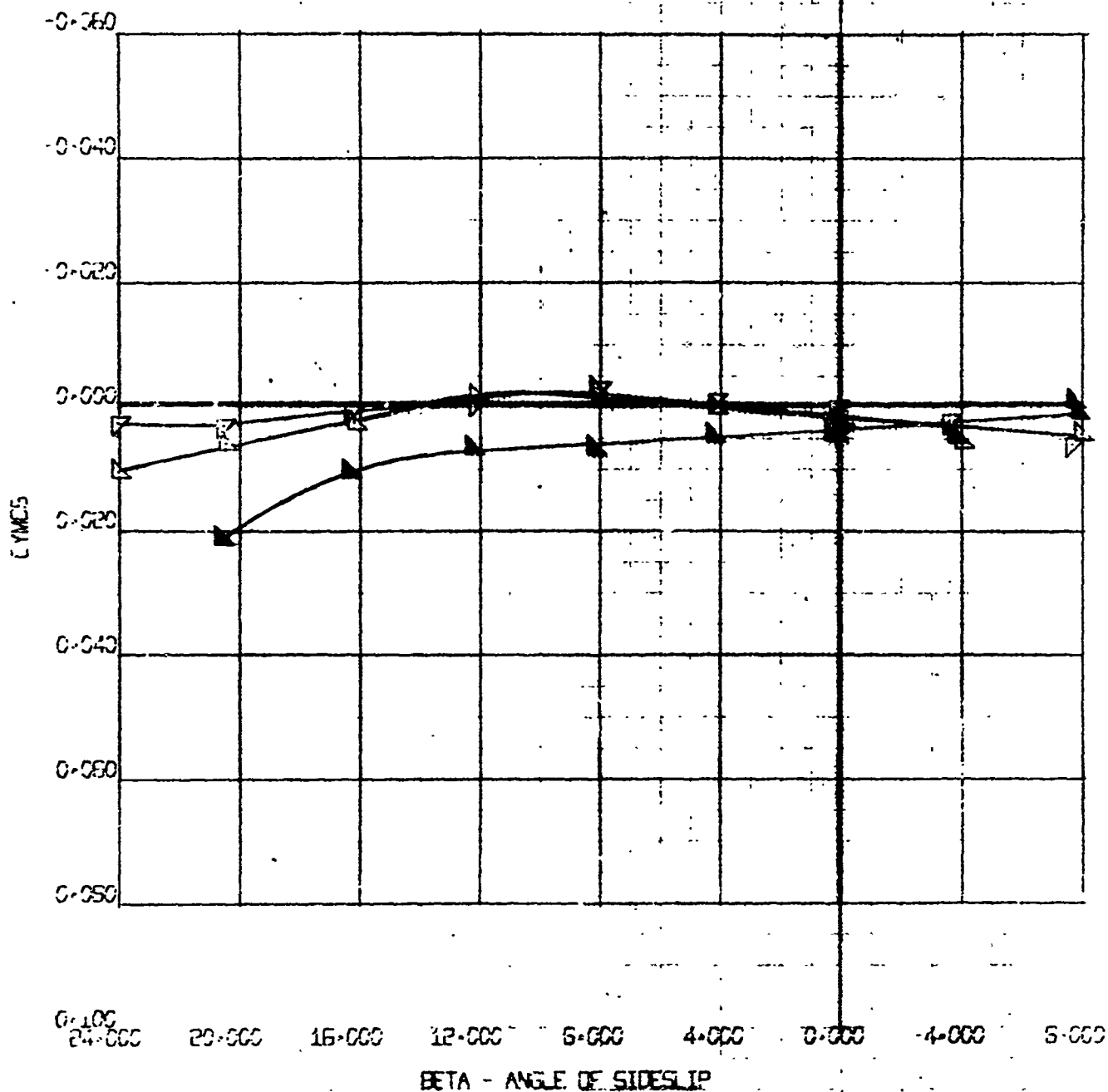
FOUR-PROP TILT WING	BVWT
MODEL VR0685 (FULL SPAN)	67
CYMCS VS. BETA	12/17/70

RUN	SYM	λ	q	NOM C_{T_s}	γ
132	▽	+10°	4.0	.70	0°
133	△	↓	6.3	.50	↓
134	◣	↓	10.6	.25	↓

0·200   Empennage On

NUMBER   D170-10039-1
REV· LTR·   Figure 184

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=15°, $δ_f$=60°, h/D=1.25
FULL SPAN SLATS

FOUR-PROP TILT WING	BVWT
MODEL VRC58C (FULL SPAN)	67
CYCS VS· BETA	12/17 '70

SHEET 264

NOT REPRODUCIBLE
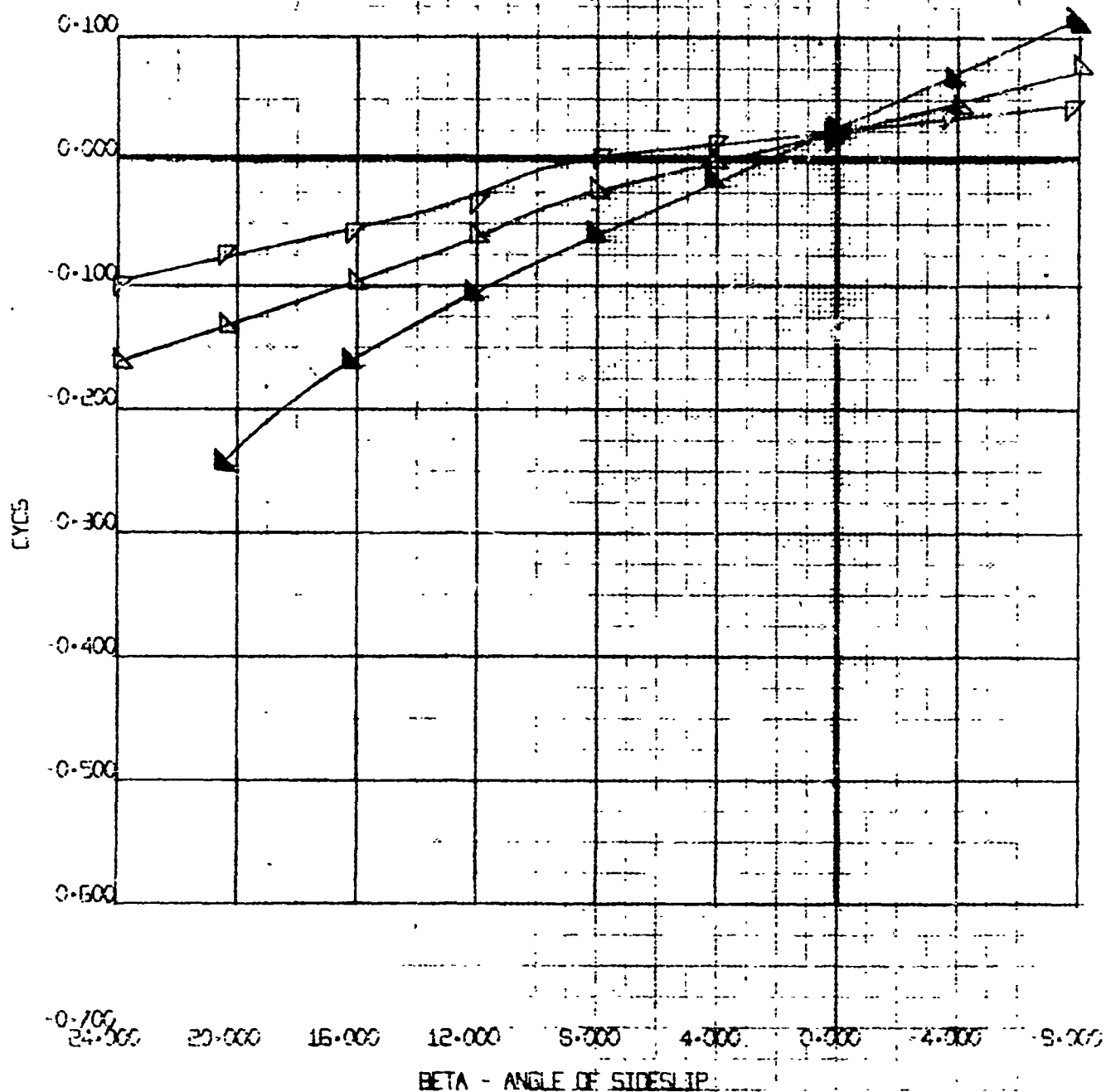
RUN	SYM	Δ	q	NOM C_{T_s}	γ
132	▽	+10°	4.0	.70	0°
133	◣	↓	6.3	.50	↓
134	◤	↓	10.6	.25	↓
0·030	Empennage On				

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=1° α=1° h/b=1.25
FULL SPAN SLATS

BETA - ANGLE OF SIDESLIP

CRMCS

FOUR-PROP TILT WING	BVWT
MODEL VROSBO (FULL SPAN)	67
CRMCS VS· BETA	12/17/70

SHEET 265

RUN	SYM	⌐	α	NOM C_{T_S}	γ
135	▽	+15°	2.7	.81	0°
136	▽		4.0	.70	
137	△		6.5	.55	

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_W=30°$ $\delta_F=60°$ $h/D=1.25$
FULL SPAN SLATS

CYMC5

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROSBG (FULL SPAN)	67
CYMC5 VS. BETA	12/17/70

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=30°$, $\delta_F=60°$, h/D=1.25
FULL SPAN SLATS

RUN	SYM	i_T	q	NOM C_{T_E}	γ
135	▽	+15°	2.7	.81	0°
136	▷		4.0	.70	
137	△		6.5	.55	

Empennage On

FOUR-PROP TILT WING
MODEL VROSEC (FULL SPAN)   BVWT 67
CYCS VS· BETA   12/17/70

SHEET 267

RUN	SYM	Δ	G	NOM C_{T_S}	γ
135	▽	+15°	2.7	.81	0°
136	▽	↓	4.0	.70	↓
137	△	↓	6.5	.55	↓

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$\lambda_w=30°$, $\lambda_F=60°$, h/D=1.25
FULL SPAN SLATS

CRMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CRMCS VS. BETA	12/17/70

SHEET 268

RUN	SYM	δ	q	NOM C_{T_s}	γ
138	$\triangledown$	+25°	2.7	.81	0°
139	$\triangleleft$	↓	1.0	.92	↓

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=45°, δ_F=60°, h/D=1.25
FULL SPAN SLATS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BWT
MODEL VRC58C (FULL SPAN)	67
CYMCS VS· BETA	12/17/70

NOT REPRODUCIBLE

SHEET 269

				NOM			NUMBER	D170-10039-1

RUN	SYM	Λ	α	C_{T_S}	γ	REV. LTR.	Figure 190
138	▽	+25°	2.7	.81	0°		
139	⊙	ᵗ	1.0	.92	ᵗ		

0.200   Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=45°, δ_F=60°, h/D=1.25
FULL SPAN SLATS

BETA - ANGLE OF SIDESLIP

CYCS

FOUR-PROP TILT WING	BVWT
MODEL VRO686 (FULL SPAN)	67
CYCS VS. BETA	12/17 '70

SHEET 270

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
i_w=45°, δ_F=60°, h/D=1.25
FULL SPAN SLATS

RUN	SYM	i	q	NOM C_T	γ
138	▽	+25°	2.7	.81	0°
139	◁	↓	1.0	.92	↓
	Empennage On				

CRMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROGBO (FULL SPAN)	67
CRMCS VS· BETA	12/17/70

SHEET 271

RUN	SYM	Δ	q	NOM C_{T_S}	ψ
158	◁	+10°	6.3	.50	-4°
163	▷	↓	↓	↓	+4°

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=15°$, $\delta_p=60°$, $h/B=1.25$
FULL SPAN SLATS

With Cyclic Pitch Control

CYMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VRC5580 (FULL SPAN)	67
CYMCS VS. BETA	1/25/71

SHEET 272

NOT REPRODUCIBLE

Lateral/Directional Stability in Ground Effect. Four-Prop Tilt Wing Model VR0680 (Full Span). CYCS vs. BETA.

FOUR-PROP TILT WING
MODEL VROSBO (FULL SPAN)
CRMCS VS. BETA

Figure 195. LATERAL/DIRECTIONAL STABILITY IN GROUND EFFECT $i_w=30°$, $\delta_F=60°$, h/D=1.25 FULL SPAN SLATS

Lateral/directional stability figure: CYCS vs. BETA for four-prop tilt wing model.

NOT REPRODUCIBLE

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=30°$, $\delta_F=90°$, $h/D=1.25$
FULL SPAN SLATS

RUN	SYM	i_w	q	NOM C_{P_s}	γ
160	◀	15°	4.0	.70	-4°
162	▷	↓	↓	↓	+4°

Empennage On

0.030

With Cyclic Pitch Control

CRMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROGBO (FULL SPAN)	67
CRMCS VS. BETA	1/25/71

RUN	SYM	l	q	NOM C_{T_s}	γ
159	▼	+15°	2.7	.81	-4°
161	▷	↓	↓	↓	+4°

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w = 30°$, $\delta_F = 60°$, h/D=1.25
FULL SPAN SLATS

-0.080

-0.060

-0.040

With Cyclic Pitch Control

-0.020

-0.000

CYMCS

0.020

0.040

0.060

0.080

0.100

24.000 20.000 16.000 12.000 8.000 4.000 0.000 -4.000 -8.000

BETA - ANGLE OF SIDESLIP

FOUR-PROP. TILT WING	BVWT
MODEL VR0680 (FULL SPAN)	67
CYMCS VS. BETA	12/21/70

SHEET 278   NOT REPRODUCIBLE

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=30°$, $i_p=60°$, $h/D=1.25$
FULL SPAN SLATS

RUN	SYM	i_p	q	NOM C_{T_S}	γ
159	▽	+15°	2.7	.81	-4°
161	▷				+4°

Empennage On

With Cyclic Pitch Control

CYCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VROG80 (FULL SPAN)	67
CYCS VS. BETA	12/21/70

NOT REPRODUCIBLE

SHEET 279

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=30°$, $\delta_F=60°$, h/D=1.25
FULL SPAN SLATS

RUN	SYM	Δ	α	C_{T_s} NOM	γ
159	▼	+15°	2.7	.81	−4°
161	▷	↓	↓	↑	+4°

0·030   Empennage On

With Cyclic Pitch Control

CRMCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING	BVWT
MODEL VRO68Q (FULL SPAN)	67
CRMCS VS· BETA	12/21/70

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=45°$, $\delta_F=60°$, $h/D=1.25$
FULL SPAN SLATS

FOUR-PROP TILT WING
MODEL VRC680 (FULL SPAN)
CYMCS VS. BETA

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=45°$, $\delta_F=60°$, h/D=1.25
FULL SPAN SLATS

RUN	SYM	δ	q	NOM C_{T_S}	γ
157	◀	+25	1.0	.92	-4°
164	◇				+4°
165	+				+6°
166	◣				-6°

Empennage On

NUMBER D170-10039-1
REV. LTR. Figure 202

With Cyclic Pitch Control

CYCS

BETA - ANGLE OF SIDESLIP

FOUR-PROP TILT WING
MODEL VR0680 (FULL SPAN)
CYCS VS. BETA

BVWT
67
12/21/70

SHEET 282

RUN	SYM	ℓ	q	NOM C_{m_s}	γ
157	◀	+25	1.0	.92	−4°
164	◇				+4°
165	+				+6°
166	◣				−6°

Empennage On

LATERAL/DIRECTIONAL STABILITY
IN GROUND EFFECT
$i_w=45°$, $\delta_F=60°$, h/D=1.25

FULL SPAN SLATS

With Cyclic Pitch Control

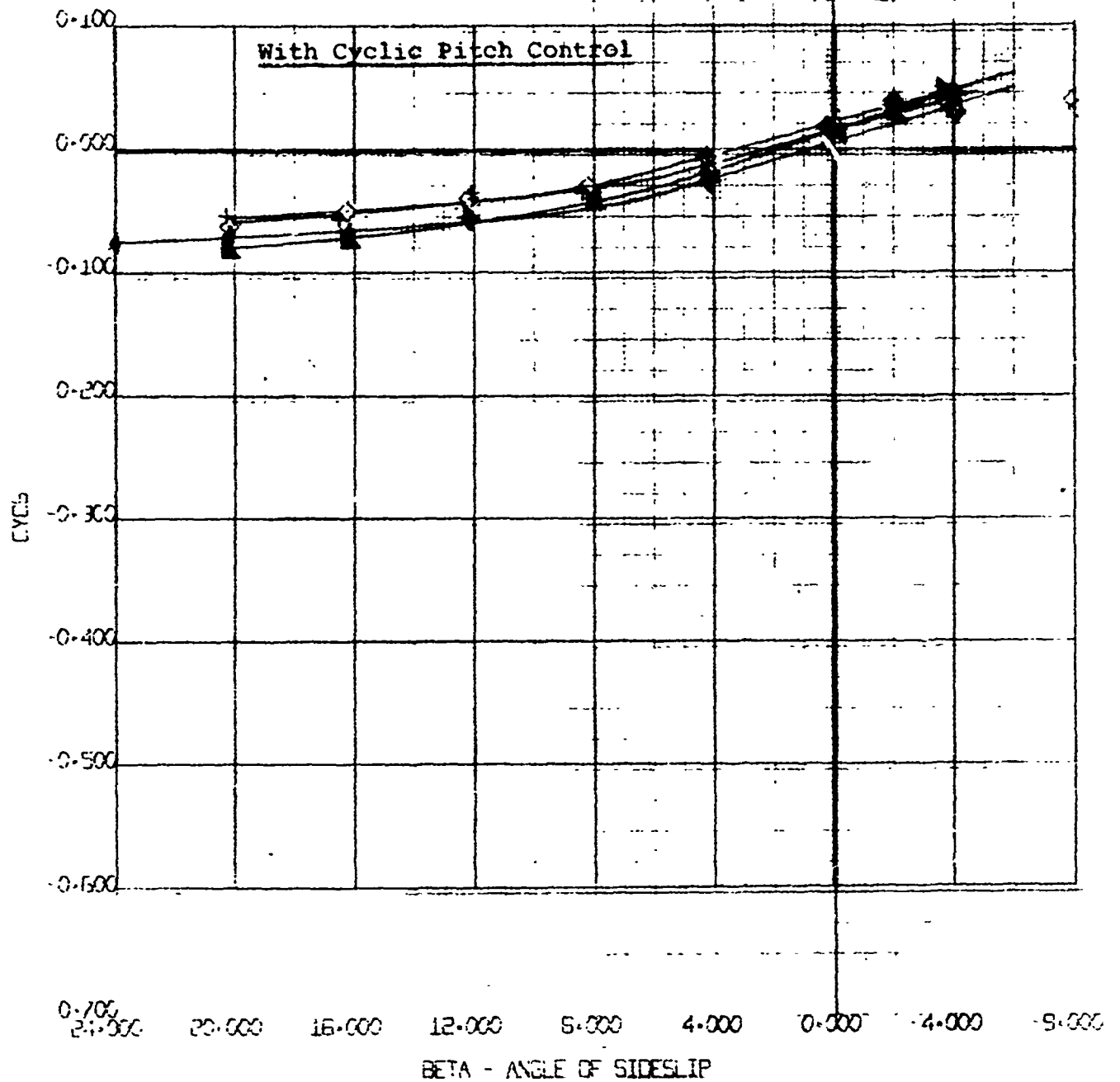BETA - ANGLE OF SIDESLIP

CRMCS

FOUR-PROP TILT WING	BVWT
MODEL YROG80 (FULL SPAN)	67
CRMCS VS. BETA	12/21/70

NOT REPRODUCIBLE

SHEET 283

7.0 CONCLUSIONS

The primary conclusions that can be derived from this Phase II test of the full span tilt wing model in hover and through the transitional flight regime are as follows:

a. Hub pitching moments due to cyclic pitch were the same in the presence of a highly flapped wing as for the isolated propeller. Total airplane pitching moments due to cyclic were greater than the sum of the hub moments, as a result of the favorable propeller normal forces generated by cyclic pitch action.

b. Pitching moments per degree of cyclic were essentially unaffected by wing tilt, forward speed or ground effect. Cyclic effectiveness was maintained up to and beyond wing stall.

c. Application of positive (nose down) cyclic reduced rate of descent capability by about 100 ft/min per degree of cyclic at 62 knots full scale speed and by about 50 ft/min per degree of cyclic at 38 knots.

d. Cyclic pitch action had only a small effect in the transition regime on the stabilizer control capability and on the roll/yaw control power from the wing surface controls, flaps and spoilers. Cyclic pitch did alter the hover yaw control power from the flaps and spoilers with a positive cyclic angle of 6° increasing the surface control power by 15% and a negative cyclic angle of 6° decreasing it by 17%.

e. Cross-coupling effects of the stabilizer and roll/yaw wing surface controls on the cyclic pitch control effectiveness were not significant.

f. The use of cyclic pitch had no marked effect on either longitudinal stability or lateral/directional stability through transitional flight or in-ground effect at a 3 foot wheel height with the exception of the increase in directional stability and dihedral effect recorded in-ground effect with negative cyclic.

g. A hover yaw control configuration of 60° down flaps on one wing and 60° up spoilers on the opposite wing appears to be the most favorable yaw control configuration of the various combined control configurations evaluated. This assertion considers both the out-of-ground effect control capability and the reduction that occurs in-ground effect (29% for the noted configuration at a 2 ft. wheel height).

h. The necessary value of the non-dimensional yawing moment parameter (Y.M./ℓT) is 0.29 for a representative tilt-wing transport type aircraft hovering at its design "V" gross weight of 87,000 lbs. and meeting a 0.5 radian/sec^2 yaw angular acceleration level. The hover yaw control configuration of combined down flaps and up spoilers produces a moment that is within 14% of the goal. This percentage difference takes into account the 6.5% increase in out-of-ground control capability that was observed when the disc loading was increased from 16.7 lb/ft^2 to 33.4 lb/ft^2

i. Hover download due to application of yaw control increased approximately linearly with yawing moment, reaching a level of 14% of thrust with 60° flaps on one side and 60° spoilers on the other.

j. At the high speed end of transition (low slipstream thrust coefficients) with the wing in the down position, the effectiveness of the vertical tail in terms of developing yawing moment was low. This problem has been identified as a flow problem and could be corrected by redesigning the wing/body juncture and the intersection between the base of the fin and the top of the fuselage.

k. In the low speed part of the transition (high slipstream thrust coefficients) with the wing tilted up to angles in the order of 45° and 55°, the aircraft was directionally stable. This situation is attributed to the observed change in the character of the flow about the fuselage at high wing tilt angles and the subsequent redistribution of longitudinal pressures on the fuselage.

285

1. A ground recirculation condition was encountered during the in-ground effect testing with the model set at an equivalent full scale wheel height of 3 ft. With the wing tilted to 30° and the flaps deflected to 60°, ground recirculation was in the initial stages of formation at 0.8 C_{T_S}. At higher C_{T_S} values with 45° of wing tilt, ground recirculation was well developed.

m. The influence of the ground on the lift curves and force polars for the 60° flap deflection was typical except for the ground recirculation condition at high C_{T_S} values. At low wing incidences, an increase in the slipstream lift coefficient wqs recorded and a decrease in the slipstream drag coefficient was generally evident. Maximum slipstream lift coefficient was decreased by a constant increment of 0.25 C_{L_S}.

n. A check of the stabilizer effectiveness in-ground effect at a C_{T_S} of 0.55 showed some increase in $\Delta C_{M_S}/\Delta\alpha$ over that recorded out-of-ground effect with comparable model conditions.

o. The major influence of the ground on the lateral/directional stability characteristics in the C_{T_S} range from 0.28 to 0.71 was to decrease the directional stability.

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